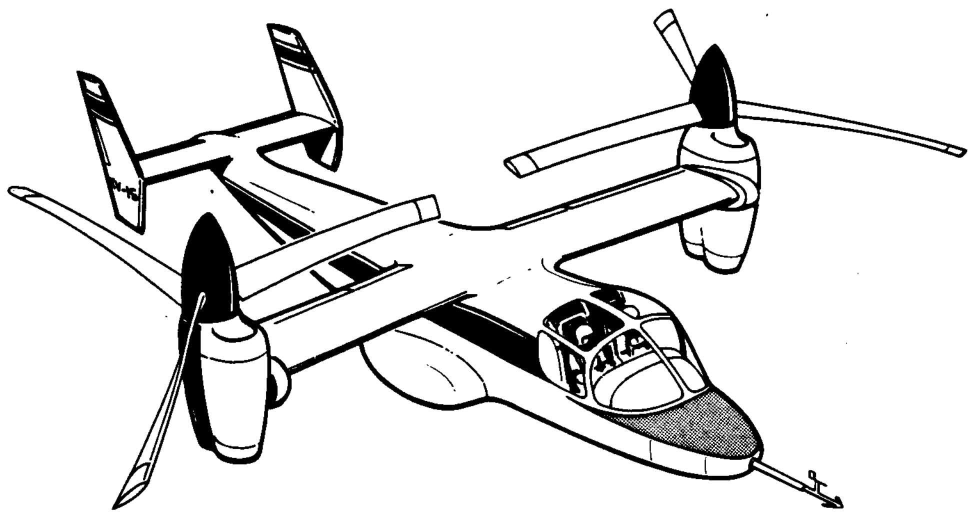
NASA TECHNICAL
MEMORANDUM

NASA TM X-62, 407

~~RELEASE BY ORIGINATING OFFICE ONLY~~

NASA TM X-62, 407



NASA/ARMY
XV-15

TILT ROTOR RESEARCH AIRCRAFT FAMILIARIZATION DOCUMENT

Prepared by Tilt Rotor Project Office Staff
Coordinated by Martin Maisel

Ames Research Center
and
U.S. Army Air Mobility R&D Laboratory
Moffett Field, Calif. 94035

January 1975

NASA/ARMY XV–15 TILT ROTOR RESEARCH AIRCRAFT

FAMILIARIZATION DOCUMENT

Prepared by: Tilt Rotor Research Aircraft Project Office Staff

Coordinated by: Martin D. Maisel

Martin D. Maisel
Tilt Rotor Research Aircraft Project Office

Approved by: Dean C. Borgman

Dean C. Borgman
Deputy Manager, Technical
Tilt Rotor Research Aircraft Project Office

David D. Few

David D. Few
Manager
Tilt Rotor Research Aircraft Project Office

1. Report No. TM X-62,407	2. Government Accession No.	3. Recipient's Catalog No.
4. Title and Subtitle NASA/ARMY XV-15 TILT ROTOR RESEARCH AIRCRAFT FAMILIARIZATION DOCUMENT		5. Report Date 6. Performing Organization Code
7. Author(s) Prepared by Tilt Rotor Project Office Staff, coordinated by Martin Maisel		8. Performing Organization Report No. A-5870
9. Performing Organization Name and Address NASA Ames Research Center and U.S. Army Air Mobility R&D Laboratory Moffett Field, Calif. 94035		10. Work Unit No. 744-01-01 11. Contract or Grant No.
12. Sponsoring Agency Name and Address National Aeronautics and Space Administration Washington, D.C. 20546		13. Type of Report and Period Covered Technical Memorandum 14. Sponsoring Agency Code
15. Supplementary Notes		

16. Abstract

The design features and general characteristics of the NASA/Army XV-15 Tilt Rotor Research Aircraft are described. This aircraft was conceived as a proof-of-concept vehicle and a V/STOL research tool for integrated wind tunnel, flight-simulation, and flight-test investigations. Discussions of special design provisions and safety considerations necessary to perform these missions are included in this report. In addition to predictions of aircraft and engine performance for the hover, helicopter, and airplane flight modes, analytical estimates of the structural and dynamic limitations of the XV-15 are provided.

17. Key Words (Suggested by Author(s)) V/STOL tilt rotor; Research aircraft Aircraft specifications and performance Structural dynamics Subsystems; Safety features		18. Distribution Statement Unclassified - Unlimited STAR Category - 05	
19. Security Classif. (of this report) Unclassified	20. Security Classif. (of this page) Unclassified	21. No. of Pages 105	22. Price* $5.25

*For sale by the National Technical Information Service, Springfield, Virginia 22151

TABLE OF CONTENTS

NASA/ARMY XV-15 TILT ROTOR RESEARCH AIRCRAFT

FAMILIARIZATION DOCUMENT

Prepared by the Tilt Rotor Project Office Staff
Ames Research Center

Coordinated by Martin D. Maisel

1. SCOPE AND INTRODUCTION

This document provides a summarization of the characteristics and features of the XV-15 Tilt Rotor Research Aircraft based on data available as the detail design nears completion. (A previous version of this document was issued early in the detail design stages.) Because of the experimental shop approach taken in the design and manufacture of the XV-15, the aircraft's configuration is not intended to be frozen until rollout. As a result, the data presented herein is subject to change (without notice to holders of this document). In addition, all predicted performance values quoted in this document are research goals and not guaranteed values. Only those design characteristics which may affect flight safety are firm requirements.

The XV-15 Tilt Rotor Research Aircraft is the product of several years of technology development carried out by the Army and NASA at Ames Research Center. In the 1950's the XV-3 flight program provided a limited demonstration of the feasibility of the tilt rotor concept at low speed and identified problems requiring further research. Subsequent investigations, including wind tunnel testing of flightworthy rotor systems, has verified the solutions to the basic rotor/pylon/wing dynamic stability problems evidenced in the XV-3. This new technology base has been applied to the current tilt rotor program.

The XV-15 research aircraft is the minimum size required to perform proof-of-concept flight investigations. In addition, it can be accommodated in the NASA/Ames Research Center 40- by 80-Foot Wind Tunnel and therefore undergo full scale testing prior to flight. First flight of the XV-15 is currently anticipated in the latter part of 1976. The initial flight investigations, which will be preceeded by training periods on the Flight Simulator for Advanced Aircraft (FSAA) at Ames Research Center, will establish the aircraft's airworthiness for a limited flight envelope. The flight envelope will be fully expanded in subsequent flights to investigate the tilt rotor aerodynamic, dynamic, structural, and environmental characteristics; disc loading variations; and aircraft handling and control qualities. It is anticipated that the wind tunnel and flight simulator facilities at Ames Research Center will be used throughout the XV-15 Flight Research Program as new areas of tilt rotor technology are explored.

Two XV-15 Tilt Rotor Research Aircraft are being designed, fabricated and tested by the Bell Helicopter Company under Contract NAS2-7800, which is managed by a joint NASA/Army Project Office team at Ames Research Center.

The Bell Helicopter Company is supported in their effort by several major subcontractors, as shown below.

Prime contractor:	Bell Helicopter Company
Major subcontractors:	
Fuselage and empennage	Rockwell International - Tulsa Division
Engines	AVCO - Lycoming
Automatic flight control system	Calspan Corporation
Conversion and flap drive systems	Steel Products Engineering Company (SPECO)

2. AIRCRAFT DESCRIPTION

The XV-15 Tilt Rotor Research Aircraft (fig. 2.1) will be representative of those aircraft which will employ the tilt rotor concept and will display generic tilt rotor characteristics as described in the following paragraphs.

The hover lift and cruise propulsive force is provided by low disc loading rotors located at each wing tip. The rotor axes rotate from the vertical (or near vertical), the normal position for hover and helicopter flight, to the horizontal for airplane mode flight. Hover control is provided by rotor generated forces and moments while airplane mode flight control is obtained primarily by the use of conventional aerodynamic control surfaces.

A cross-shafting system connecting the rotors provides several benefits. This system precludes the complete loss of power to either rotor due to a single engine failure, permits power transfer for transient conditions, and provides rotational speed synchronization. Rotor axis tilt synchronization is achieved by a separate interconnect shaft.

The aircraft is capable of high duration hover (approximately one hour at design gross weight), helicopter mode flight, versatility in performing conversion (steady state flight is possible at any point in the broad transition corridor), and airplane mode level flight at speeds greater than 300 knots. The low disc loading rotors can be operated in the autorotation state to reduce descent rate in the event of total power loss. Research operation at the design gross weight allows for a total useful load of over 2900 pounds.

At intermediate rotor axis tilt angles (between 60° and 75°) the aircraft can perform STOL operations at weights above the maximum VTOL gross weight.

The Lycoming LTC1K-4K engines (a modification of the T53-L-13B) and main transmissions are located in wing-tip nacelles to minimize the operational loads on the cross-shaft system and, with the rotors, tilt as a single unit. The use of the free turbine engines permits the reduction of rotor rotational speed for airplane mode flight to improve rotor performance and reduce cruise noise.

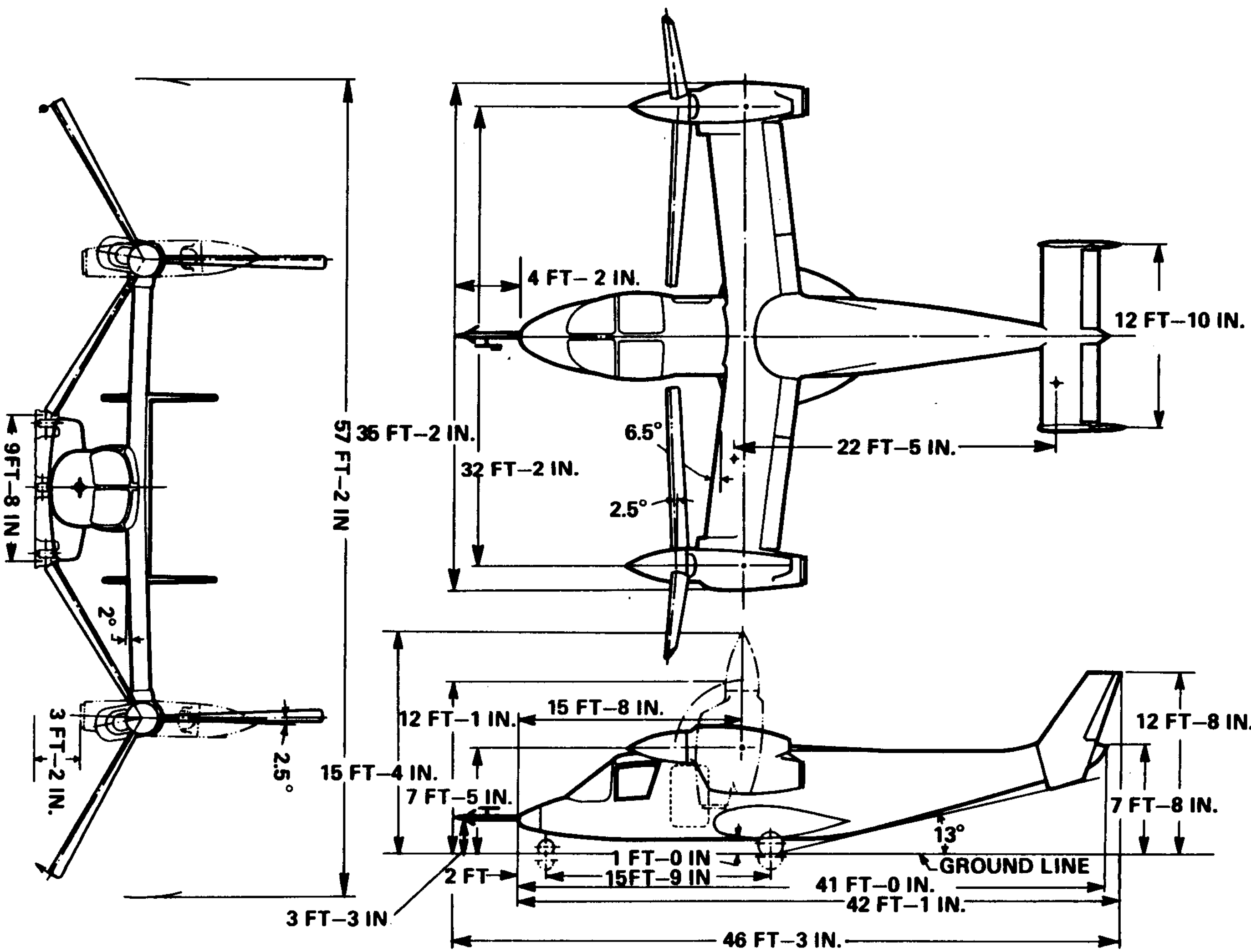


Figure 2.1.- XV-15 Tilt Rotor Research Aircraft.

The Tilt Rotor Research Aircraft utilizes 25-ft-diam gimbal-mounted, stiff-inplane, three-bladed rotors, with elastomeric flapping restraints for increasing helicopter mode control power and damping. The forward swept wings provide clearance for the 12° of flapping which will accommodate gusts and maneuver excursions while operating in the airplane mode. Wing/rotor/pylon stability is accomplished by selecting a stiff wing and pylon-to-wing attachment and minimizing the distance of the rotor hub from the wing. Airplane mode wing/rotor/pylon stability is retained up to airspeeds of 370 knots even with a 20 percent reduction in wing and pylon stiffness.

For hover flight, the wing flaps and flaperons are deflected downward to reduce the wing download and increase hovering efficiency. Hover roll control is provided by differential rotor collective pitch, pitch control by cyclic pitch, and yaw control by differential cyclic pitch. Pilot controls in the helicopter mode are similar to that of a conventional helicopter. A collective stick provides power and collective pitch for height control and a control stick provides longitudinal and lateral control.

In the airplane mode, conventional airplane stick and rudder pedals are employed while the collective stick/power lever continues to be used for power management. An H-tail configuration (two vertical stabilizers) has been selected to provide improved airplane directional stability around a zero yaw angle. Control authority for the power lever, blade pitch governor, cyclic, differential cyclic, differential collective, and flap/flaperon relationship are phased with mast angle by mechanical mixing linkages.

The structure of the fuselage and empennage is of conventional semi-monocoque design. External loads from the wing, landing gear, and empennage are distributed into the skin by major frames and four main longerons. Between frames, the skin is stiffened by rings and stringers. A bulkhead between cockpit and cabin supports the ejection seats for the crew. The horizontal and vertical stabilizers have single-cell torque boxes consisting of front and rear spars and stiffened upper and lower skin panels for the primary structure. Intermediate ribs are provided at movable surface hinge points. Access doors are provided for control system maintenance and to facilitate assembly of the vertical stabilizers to the horizontal stabilizer. The elevator and rudders are of conventional spar, rib, and skin construction.

Safety is of paramount importance in the design of the Tilt Rotor Research Aircraft. The size of the Tilt Rotor Research Aircraft itself contributes to its safety. While the aircraft is large enough to accomplish the objective of the project, i.e., demonstrate proof-of-concept, it is also small enough for full-scale testing in the Ames 40- by 80-Foot Wind Tunnel prior to first flight. This feature can also be profitably exploited in later advanced research programs. Additional safety provisions include pilot and copilot zero-zero ejection seats and redundant, fail operational critical aircraft systems and components. No single subsystem failure will result in a critical unsafe condition and, with automatic indication of the failure on the crew advisory light panel, normal flight operations may be continued. No double failure will prevent the crew from exercising the option of ejection from the aircraft. Rotor blade and hub components, as well as the transmission cases, are recognized exceptions to this requirement, and special conservative design considerations and adequate testing will be applied to establish that the probability of failure for these elements is negligible.

The general arrangement of the XV-15 aircraft and the placement of key subsystem components described in this document are shown in the inboard profile drawings, figures 2.2 and 2.3.

3. AIRCRAFT SPECIFICATIONS

Some of the data presented herein are subject to change as final design and manufacturing are accomplished.

The weight of equipment and installations required for tilt rotor research aircraft missions which are delivered as part of the XV-15 aircraft are contained in the research mission empty weight. This group includes the oxygen installation, research instrumentation equipment, the environmental control unit, and certain avionics and navigation equipment. The crew, fuel, ballast, oil and additional research instrumentation not initially provided with the aircraft are considered to be useful load.

In evaluating group weights or aircraft weight fractions it must be recognized that the XV-15 has many built in features for safety and research that impose weight penalties that are not intrinsic to the concept. This category includes the ejection seat and attaching hardware, the crashworthy fuel cells, fail safe redundant critical flight control elements, high crash load factors, and an oxygen system. Appropriate adjustments must be made when comparing the following weight data to aircraft not designed and equipped for similar flight research.

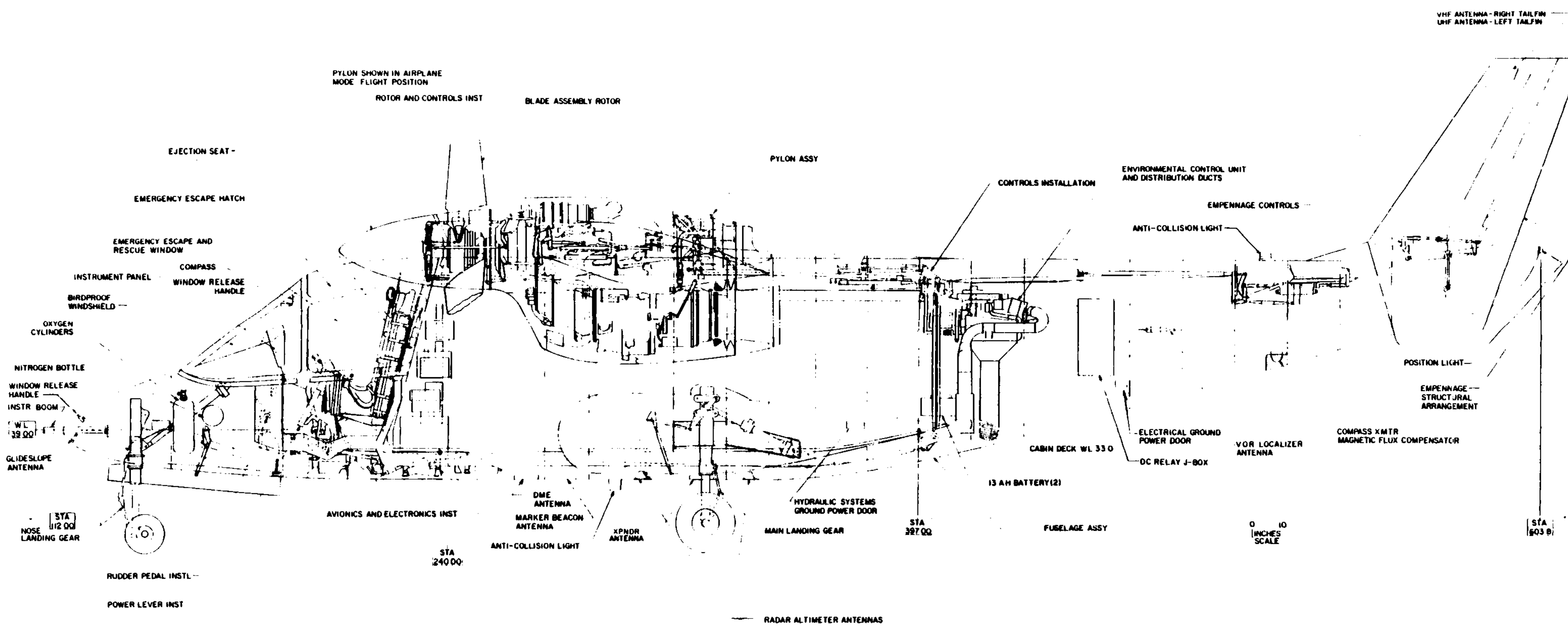
3.1 Weight and Inertia

3.1.1 *Aircraft weights, lb (as of Nov. 1974) -*

Design gross	13,000
Maximum gross	15,000
Basic empty weight	9,076
Research mission empty weight	10,073

3.1.2 *Group weights, lb*

Rotor	1,070
Wing	873



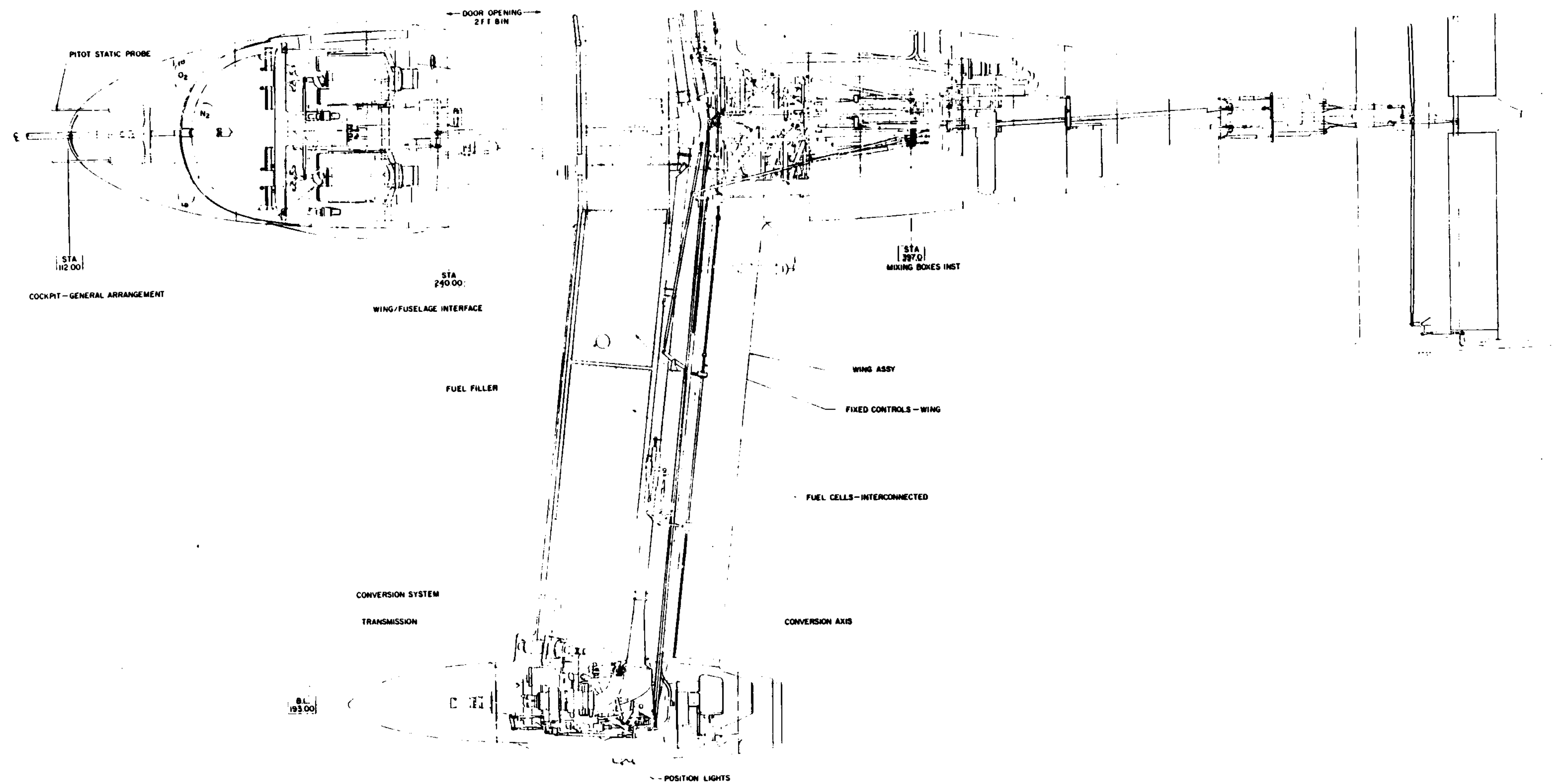
ORIGINAL PAGE IS
OF POOR QUALITY

Figure 2.2.- XV-15 inboard profile, side view.

PRECEDING PAGE BLANK NOT FILMED

FOLDOUT FRAME 1

FOLDOUT FRAME 2



ORIGINAL PAGE IS
OF POOR QUALITY

Figure 2.3.- XV-15 inboard profile, plan view.

PRECEDING PAGE BLANK NOT FILMED

FOLDOUT FRAME 1

FOLDOUT FRAME 2

Tail		209
Fuselage		1,442
Alighting gear		508
Hydraulics and flight controls		934
Powerplant		1,754
Transmission/conversion systems		1,263
Heating, air conditioning		100
Electrical		396
Instrumentation		91
Miscellaneous (furnishings, equipment)		436
Basic empty weight		9,076
Additional equipment		997
Oxygen system	60	
Fixed research instrumentation	300	
Portable research instrumentation	449	
Avionics and navigation	144	
Environmental control package	27	
Shaker	17	
Research mission empty weight		10,073
Useful load at design gross weight		2,927
Crew (2)	400	
Fuel (maximum)	1,490	
Oil and trapped fluids	138	
Additional payload	899	
Design gross weight		13,000

3.1.3 *Inertias (slug - ft², 13,000 lb gross weight)* -

	Airplane	Helicopter
I_{XX} (Roll)	40,500	42,400
I_{YY} (Pitch)	13,200	14,300
I_{ZZ} (Yaw)	50,300	49,500

3.2 Center of Gravity

The center of gravity excursions for the Tilt Rotor Research Aircraft are shown in figure 3.2.1.

3.2.1. *Ballast system* - Figure 3.2.1.1 shows ballast stowage provisions used to attain variations of gross weight and center of gravity location.

3.3 Dimensions and General data

	Wing	Horizontal Tail	Vertical Tail
Area, sq ft	169.0[a]	50.25	50.5
Span, ft	32.17[a]	12.83	7.68
Chord, ft	5.25	3.92	4.09/2.40[b]
$\bar{c}$, ft[c]	5.25	3.92	3.72[d]
Aspect ratio	6.12	3.27	2.33
Incidence, deg	3.0	0 to +6[f]	---
Dihedral, deg	2.0	0	---
Sweep $\frac{c}{4}$, deg	-6.5	0	31.6[e]
Section (NACA)	64A223 (Mod)	64A015	0009
Tail length, ft	---	22.4	23.2[d]

[a]between rotor centerlines at tilt axis; [b]root/tip; [c]mean aerodynamic chord (MAC); [d]local; [e]upper; [f]hinged @ 33 percent chord.

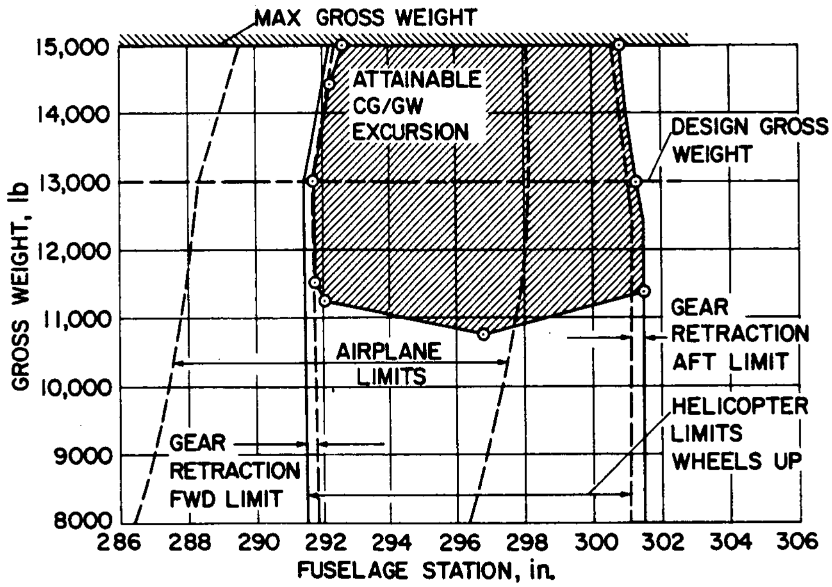


Figure 3.2.1.- Center of gravity limitations.

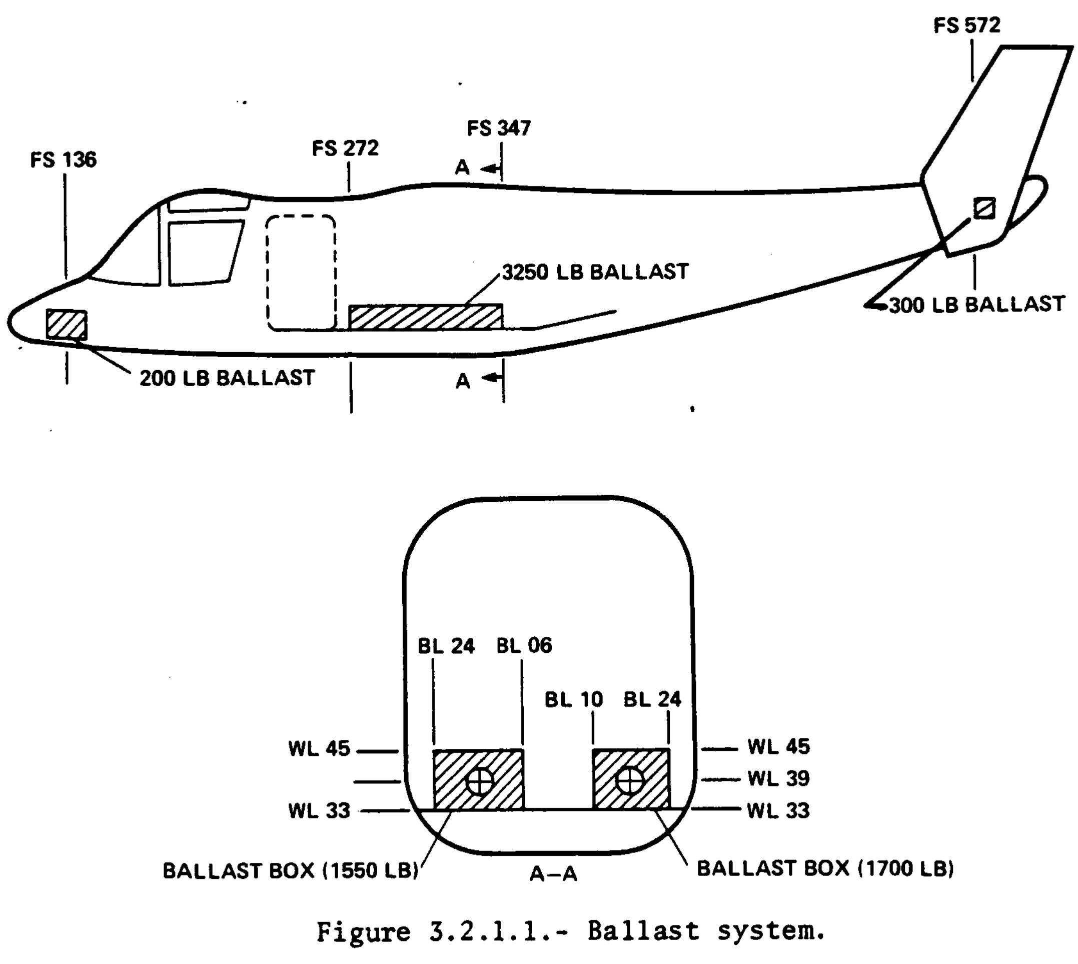


Figure 3.2.1.1.- Ballast system.

3.4 Dimensions of Movable Aerodynamic Surfaces

	Flap	Flaperon	Elevator	Rudder
Area aft hinge, sq ft	11.0[a]	20.2[a]	13.0	7.5[a]
Span, ft	4.25[b]	7.86[b]	11.0	4.66
Chord aft hinge, percent	0.25	0.25	0.30	0.25
Surface deflection, deg	---[c]	---[c]	±20	±20
Control travel for total surface deflection, in.	---	9.6	9.6	5.0

[a]total both panels

[b]one side, along hinge

[c]see figure 3.4.1.

3.5 Fuselage Cabin

Interior dimensions:

Length, ft	13.1
Width, maximum, ft	5.0
Width, at floor level, ft	4.0
Height, maximum, ft	5.0
Height, underwing, ft	4.5
Volume, cu ft	300
Floor area, sq ft	52
Usable cabin space	See figure 3.5.1

Door (right side only):

Width, ft	2.67
Height, ft	4.33

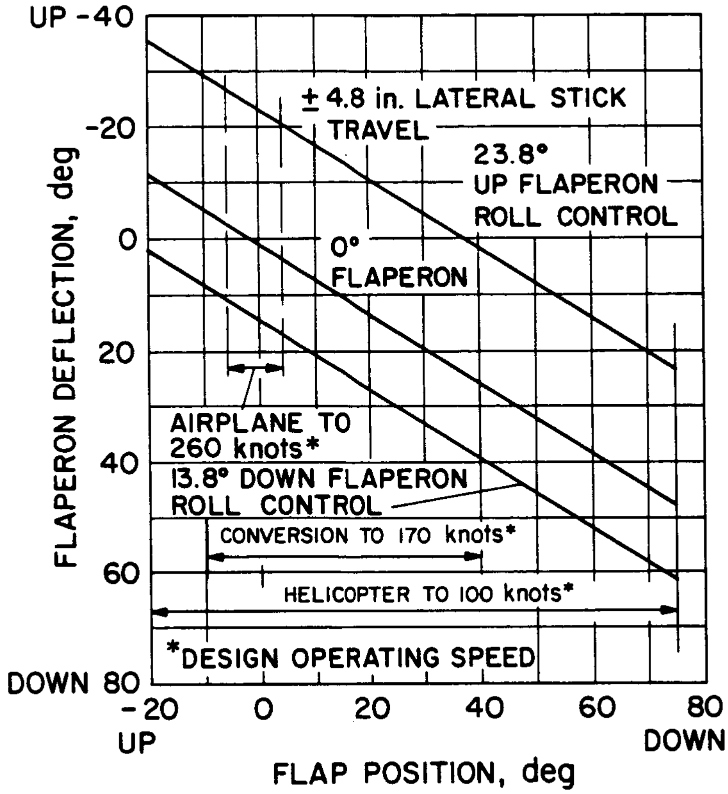


Figure 3.4.1.- Flaperon deflection versus flap position.

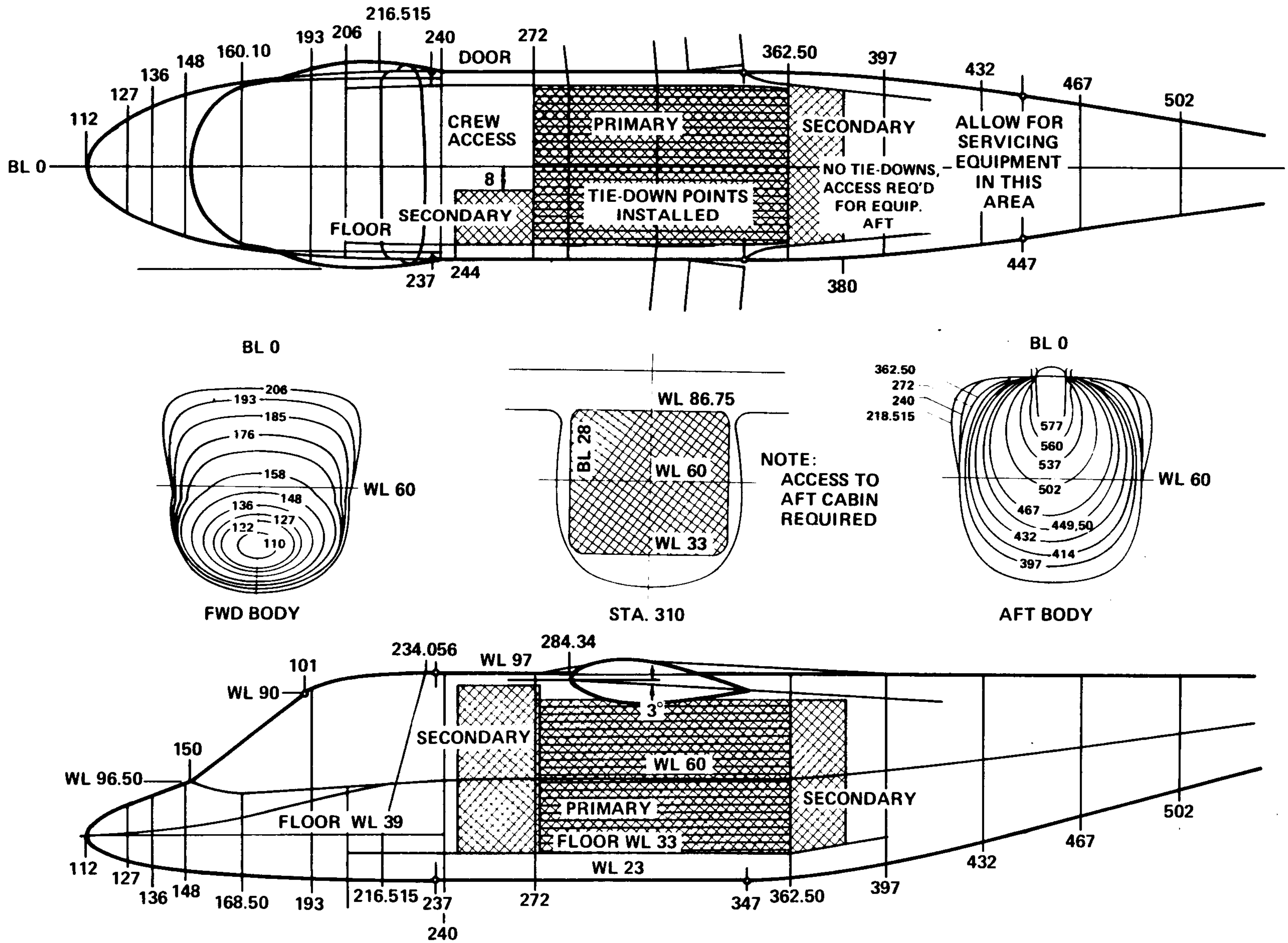


Figure 3.5.1.- Useable cabin space.

3.6 Landing Gear

Main landing gear:

Number of wheels per side	2
Wheel spacing (dual), in.	13.5
Tire size type and ply rating	7.00-8, Type III, 6 ply
Inflation pressure, psi	55
Nominal outside diameter, in.	20.5
Loading rating, lb	3250
Flat tire radius, in.	5.6
Oleo strut stroke (total), in.	13.5

Nose landing gear:

Number of wheels	2
Wheel spacing (dual), in.	11.8
Tire size type and ply rating	5.00-4, Type III, 6 ply
Inflation pressure, psi	55
Nominal outside diameter, in.	13.0
Load rating, lb	1600
Flat tire radius, in.	3.6
Oleo strut stroke (total), in.	14.5

3.7 Rotor

Number of blades per rotor	3
Diameter, ft	25.0
Disc area per rotor, sq ft	491
Blade chord (see fig. 3.7.1), in.	14.0
Blade characteristics	See figure 3.7.1
Solidity	0.089
Hub precone angle, deg	2.5
δ_3, deg	-15.0
Flapping design clearance, deg	±12
Blade Lock number	3.83

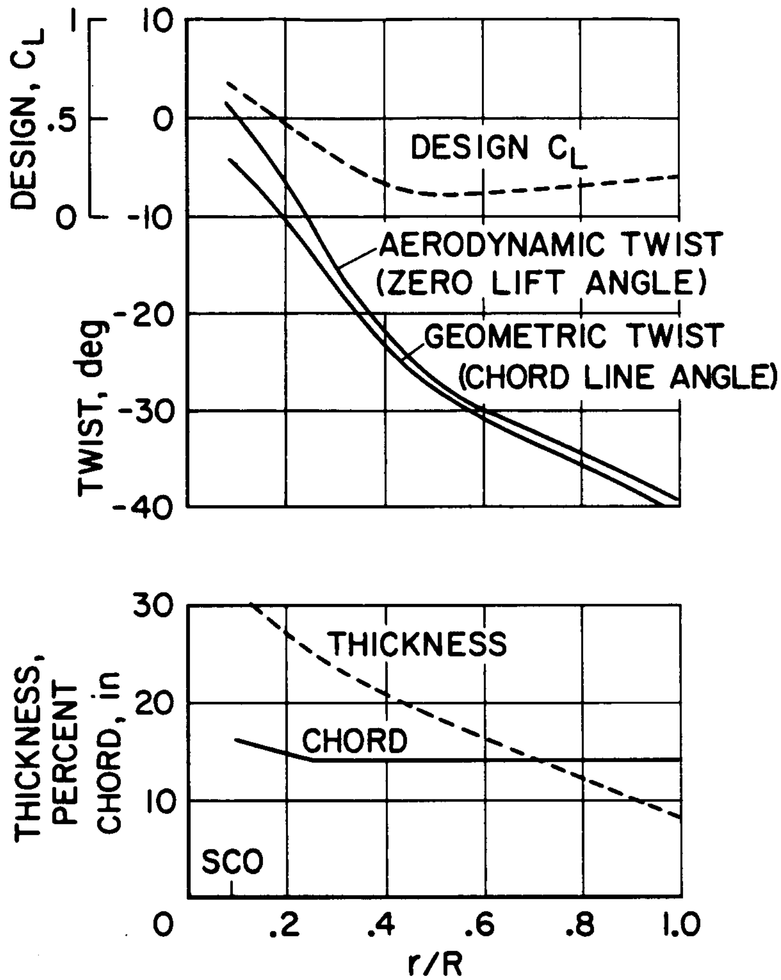


Figure 3.7.1.- Rotor blade characteristics.

Conversion axis:

Wing chord location, percent	39.0
Forward sweep, deg	5.5
Dihedral, deg	3.0
Angle-to-mast axis, deg	95.5

Angle of outboard tilt of mast axis

Helicopter mode, deg	2.5
Airplane mode, deg	0

Conversion range, deg 0 to 95

Tip clearance (helicopter) to:

Ground, ft	11.5
Wing upper surface, ft	4.8

Tip clearance (airplane) to:

Fuselage, ft	1.0
Wing leading edge, ft	0.47

3.8 Tip Speed

Variable rotor-tip speed control is provided to enable research on noise, performance, and hover downwash. Figure 3.8.1 (a and b) depicts the hover gross weight and cruise speed limitations with tip speed. The nominal design tip speeds are tabulated as follows:

Condition	Tip Speed, ft/s	rpm
Hover/Helicopter Mode	740	565
Cruise/Airplane Mode	600	458
Hover Test Overspeed	818	625
Design Limit	865	661

3.9 Loadings

The disc, blade and wing loading variations, with gross weight, are shown in figure 3.9.1.

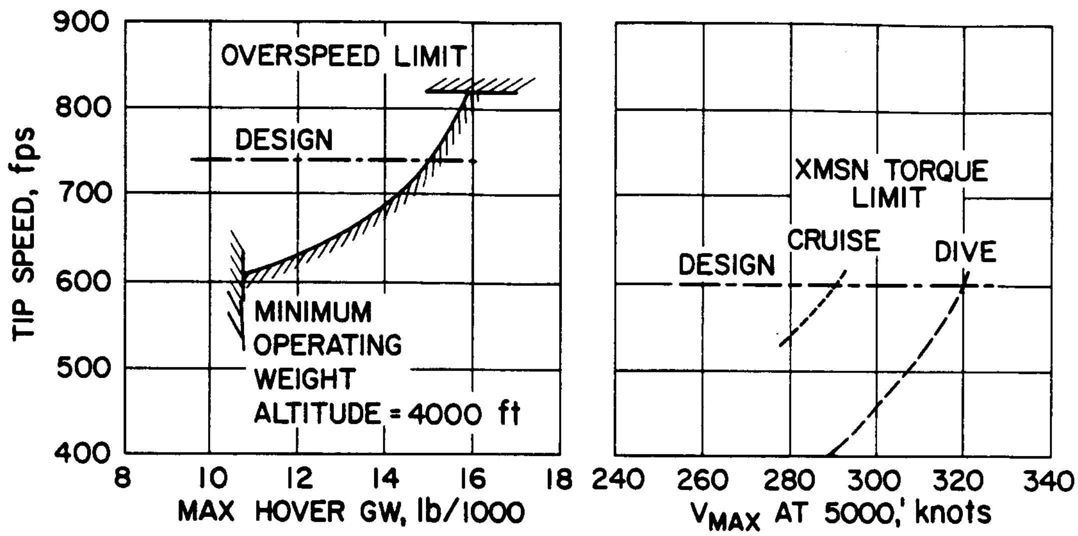


Figure 3.8.1.- Tip speed range.

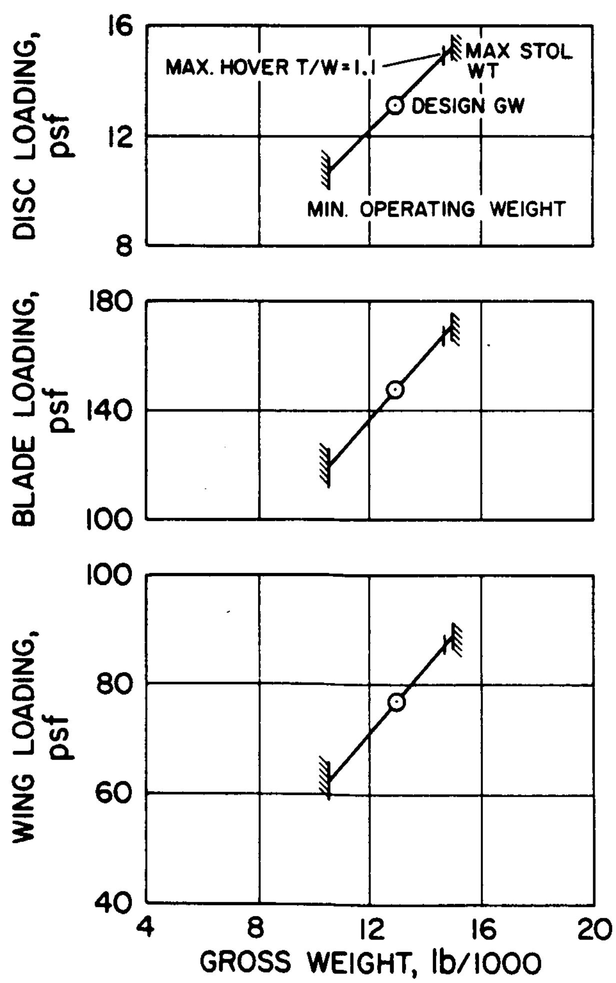


Figure 3.9.1.- Loading variation.

3.10 Ground Interference Limits

Figure 3.10.1 illustrates the ground clearance of the rotor tip for various rotor-mast-tilt and aircraft-roll angles. For this illustration, the oleo and tire on the low side of the roll are statically deflected.

4. STRUCTURAL DESIGN CRITERIA

4.1 Design Speeds and Mach Numbers

Various design structural limit speeds and Mach numbers are listed below:

Helicopter and Conversion Modes	See figure 4.1.1
Airplane Mode:	
Below 12,700 ft	300 keas
Above 12,700 ft	M = 0.575
Landing Gear Down	160 knots

4.2 Limit Load Factors and Airspeeds

The flight limit load factors and airspeeds at the minimum flying weight, the design gross weight, and maximum gross weight for the helicopter, conversion, and airplane modes are shown in figures 4.2.1 and 4.2.2.

4.3 Structural Load Factors

Selected structural load factors are presented in table 4.3.

4.4 Service Life

The Tilt Rotor Research Aircraft is designed for a minimum of 1000 flight hours over a five year period. Service life of the key aircraft elements is as follows:

Airframe:	5000 hrs
Transmissions:	3000 hrs
Rotor:	1500 hrs

4.5 Design Torques

The steady and limit torques and rotational speeds for drive system elements in various operating modes are presented in table 4.5.

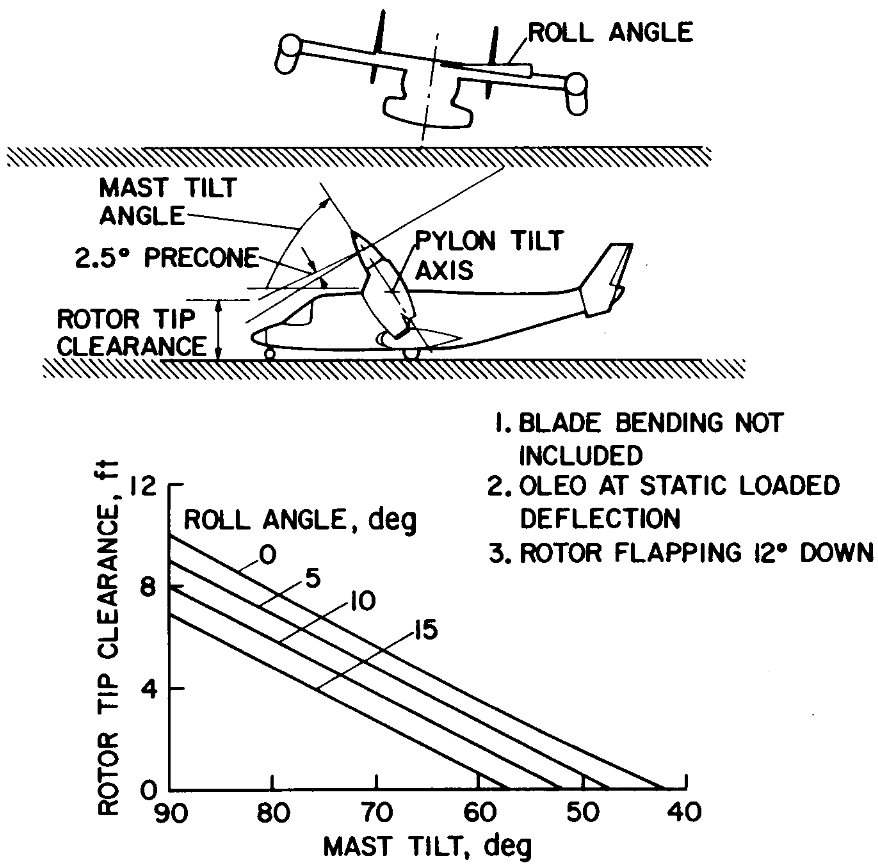


Figure 3.10.1.- Ground interference.

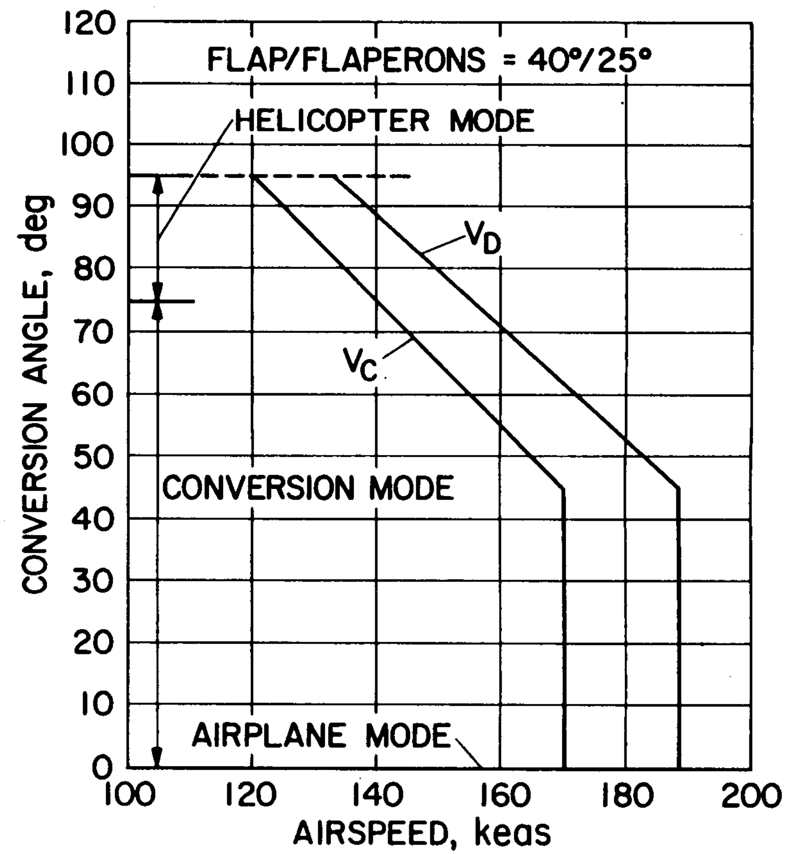


Figure 4.1.1.- Helicopter and conversion mode limit design speeds.

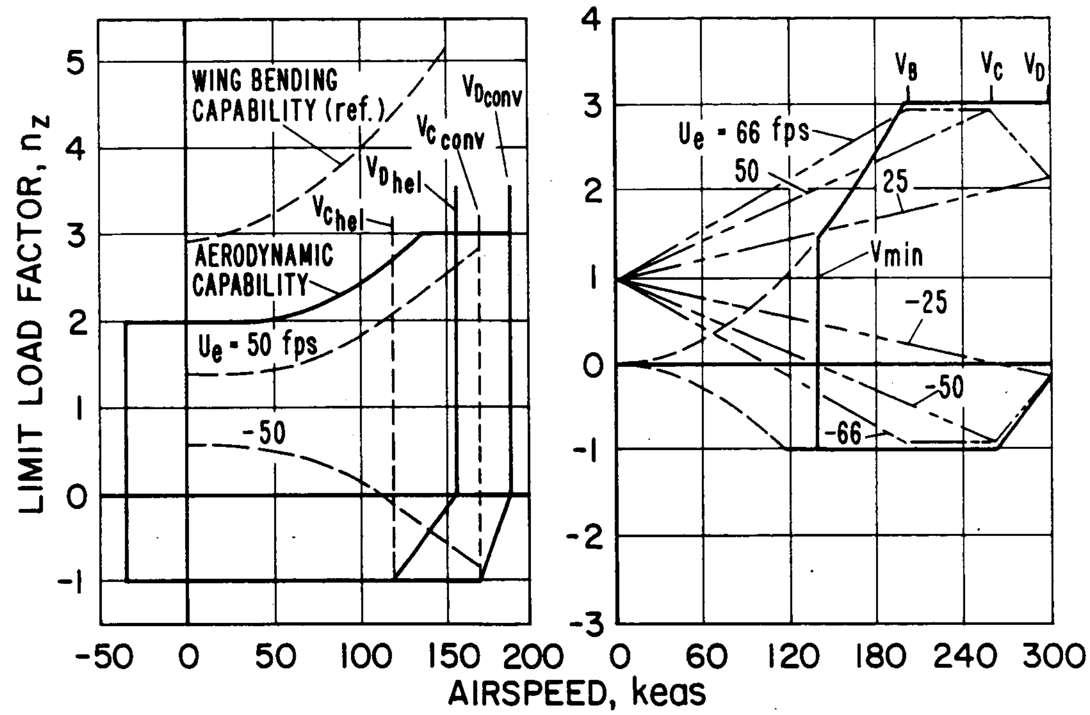


Figure 4.2.1.- V-n diagram, design gross weight, 13,000 pounds.

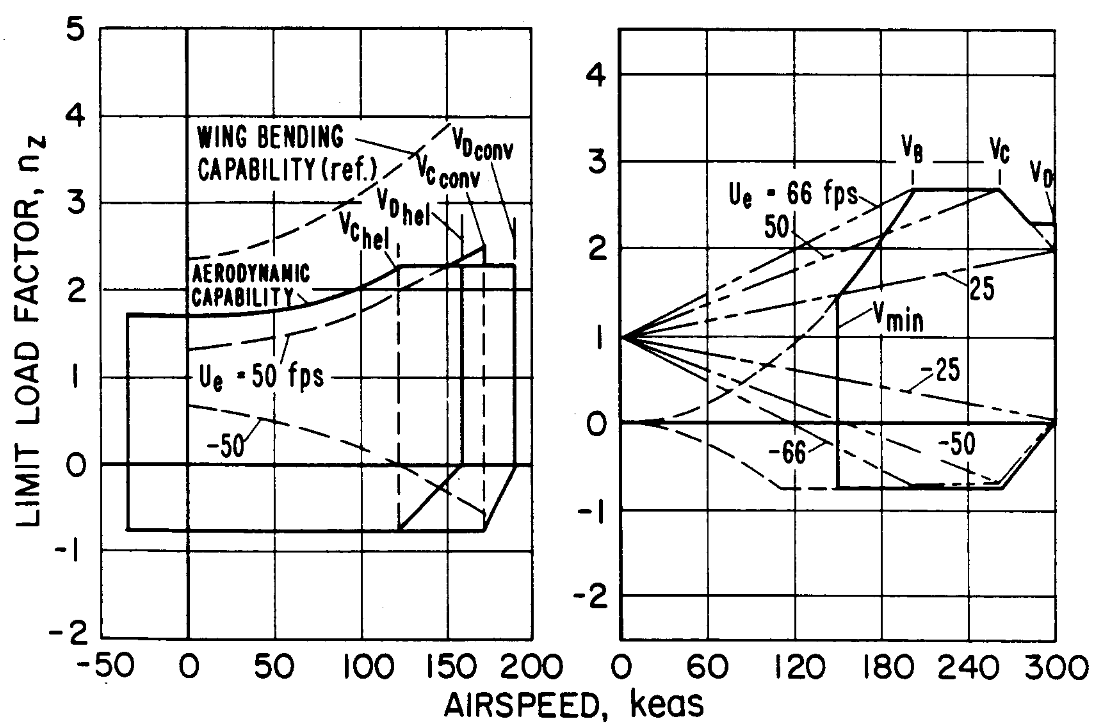


Figure 4.2.2.- V-n diagram, maximum gross weight, 15,000 pounds

TABLE 4.3.- XV-15 STRUCTURAL LOAD FACTORS

Item (FAA standard)	Factor
Ultimate factor of safety (xx.303)	1.50
Casting factor, limit (xx.619, xx.621)	1.15
Casting factor, ultimate (xx.619, xx.621)	1.25
Fitting factor (xx.619, xx.625)	1.15
Control surface aerodynamic hinge moment	1.25
Landing reserve energy	1.50

Crash load factors (ultimate):

Direction	Cockpit	Cabin		Wing-fuselage attachment		Rotor, pylon, and pylon support structure	
		Ballast boxes	Other mass items	Helicopter	Conversion and airplane	Helicopter	Conversion and airplane
Forward	15 g	9 g	15 g	15 g	15 g	7 g	7 g
Downward	23 g	4.5 g	4.5 g	15 g	10 g	15 g	10 g
Upward	5 g	2 g	2 g	5 g	5 g	5 g	5 g
Sideward	10 g	2 g	2 g	10 g	10 g	8 g	8 g

All crash load factors act separately at the center of gravity of each item.

TABLE 4.5.- XV-15 DRIVE SYSTEM

Item/flight mode	Gear ratio engine/shaft	Maximum tipspeed (fps/rpm)	Engine maximum power (shp)	Maximum steady torque (in.-lb)	Transient torque factor	Limit torque (in.-lb)
Rotor Mast	35.11:1					
Helicopter and STOL Takeoff						
Design Operating		740/565	1,460	163,000	1.33	217,000
Operating overspeed		787/601	1,460	153,000	1.33	204,000
Hover Test Overspeed		818/625	1,460	147,000	1.33	195,000
Design Limit Speed		865/661	1,460	139,000	1.33	185,000
Helicopter and Conversion, Cruise and Climb						
Design Operating		740/565	1,166	130,000	1.67	217,000
Operating Overspeed		787/601	1,240	130,000	1.67	217,000
Design Limit Speed		865/661	1,364	130,000	1.67	217,000
Airplane Cruise		600/458	946	130,000	1.67 -0.50	217,000 -65,000
Airplane Dive		600/458	1,182	163,000	1.33 -0.40	217,000 -65,000
Transmission Input Shaft	0.662:1					
One Engine Out						
Helicopter and Conversion		740/30,000	1,650	3,470	1.25	4,340
Airplane		600/24,300	1,340	3,470	1.25	4,340
Wing Interconnect Shaft	3.10:1					
One Engine Out						
Helicopter and Conversion		740/6,390	825	8,140	1.67	13,600
Airplane		600/5,180	670	8,140	1.67	13,600

5. PERFORMANCE AND NOISE

Several curves selected to illustrate the predicted performance characteristics of the Tilt Rotor Research Aircraft are presented here.

5.1 Hover

The variations of hover gross weight with altitude for the standard and 95° F (35° C) days are shown in figures 5.1.1 and 5.1.2. The performance illustrated accounts for wing download losses (7 percent loss of thrust with the flaps and flaperons deflected) and represent a useful thrust-to-weight ratio (U_T/W) of 1.0. Both normal operation (two engines) and one engine inoperative conditions are presented. Minimum operating weights show the limitations of the single engine hover capability. All predicted hover performance presented in this section is based on a transmission efficiency (η_{xmsn}) of 0.93 and rotor performance data obtained from whirl tests.

5.2 Hover Useful Load

Figure 5.2.1 depicts the useful load/altitude performance for two engine, standard day, operation. In addition to a 1.0 useful thrust-to-weight ratio, a 10 percent margin line (U_T/W = 1.1) is included to represent a conservative flight research restraint.

5.3 Hover Endurance

The duration of hover-mode flight is dependent on the quantity of fuel carried and the fuel consumption rate. Figure 5.3.1 presents an estimation of the variation of hover endurance with payload. This figure is based on the engine manufacturer's estimated specific fuel consumption (sfc) while operating at power levels corresponding to a useful thrust-to-weight ratio of 1.1 and 1.0. Hover time with 10 percent of the total fuel remaining is also shown on this figure.

5.4 Conversion Corridor

The minimum and maximum airspeed boundaries estimated for the transition process (tilt rotor mode) are shown on figure 5.4.1. The boundaries are soft and may be penetrated slightly without encountering adverse or dangerous handling characteristics, loads, or aeroelastic instabilities. The V_C and V_D limits at 170 and 189 knots respectively, are encountered due to the flap/flaperon deflections. The power required during conversion, compared to the takeoff power level is depicted in figure 5.4.2.

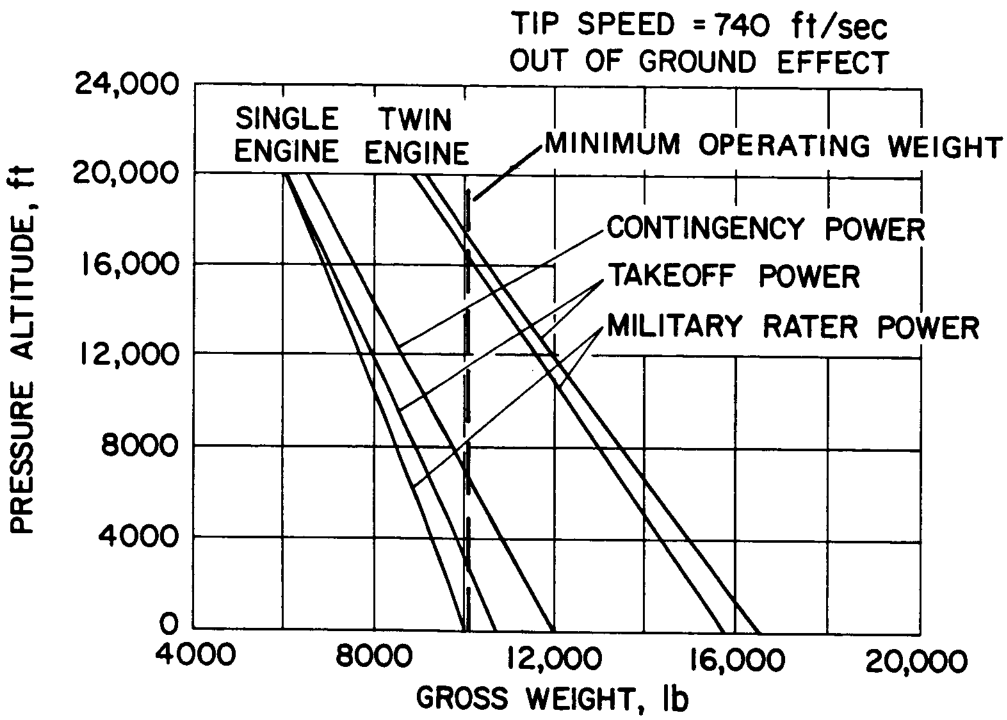


Figure 5.1.1.- Hover ceiling, standard day, ($U_T/W = 1.0$).

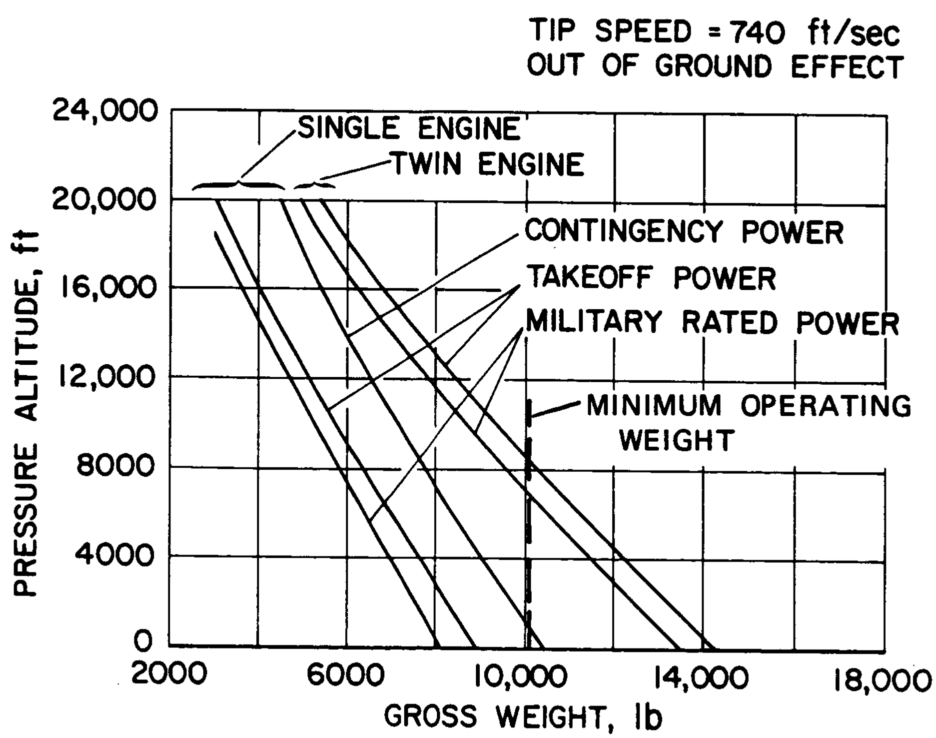


Figure 5.1.2.- Hover ceiling, T = 95° F, ($U_T/W = 1.0$).

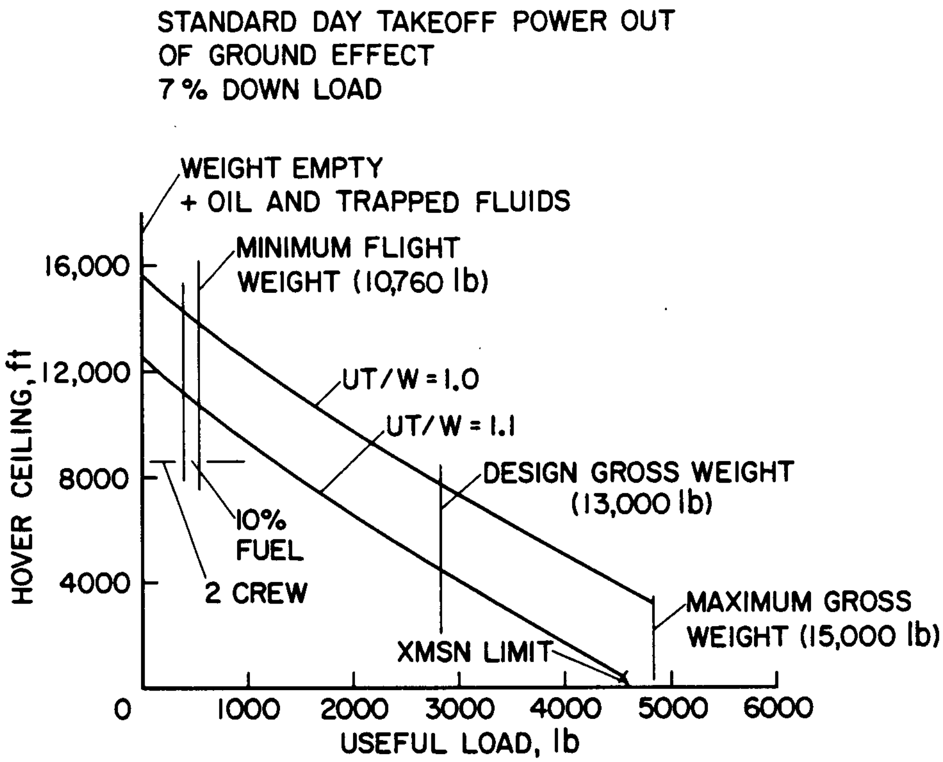


Figure 5.2.1.- Vertical lift capability.

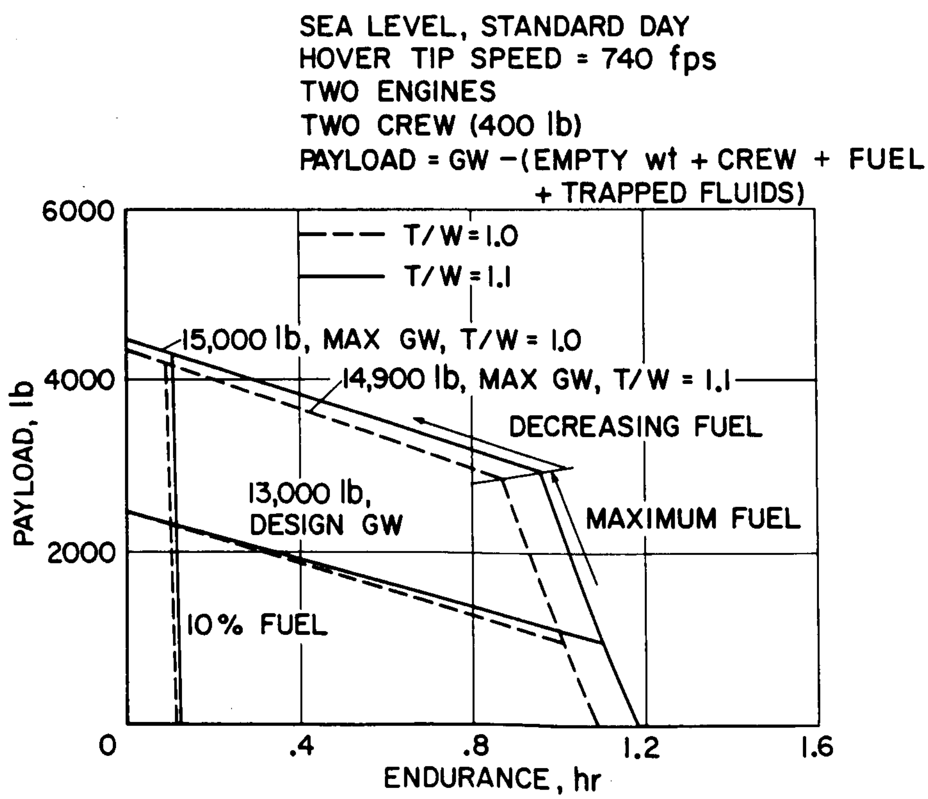


Figure 5.3.1.- Hover endurance variation with payload.

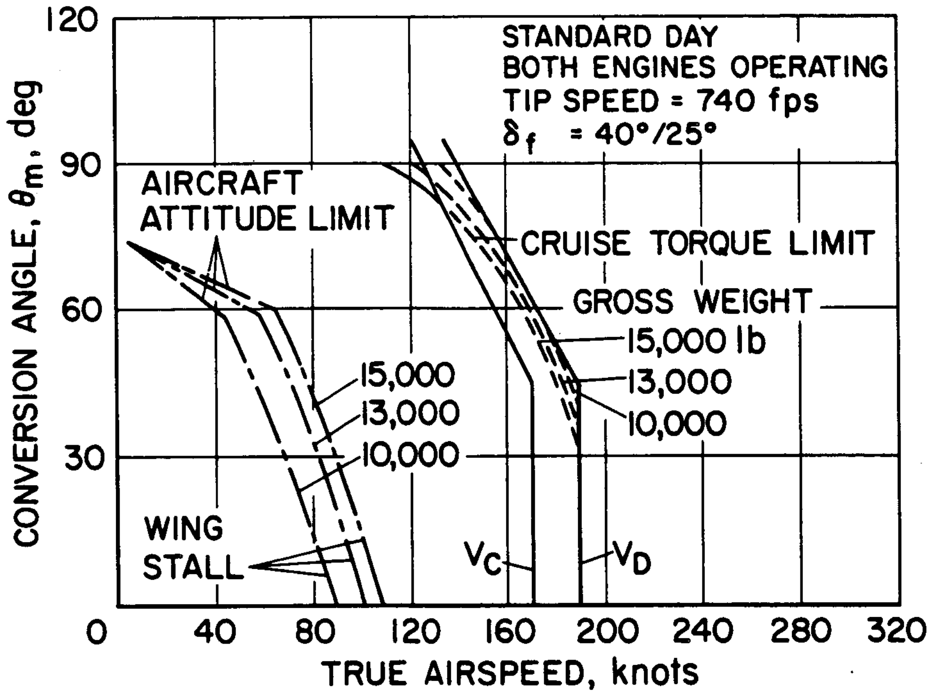


Figure 5.4.1.- Conversion corridor, sea level.

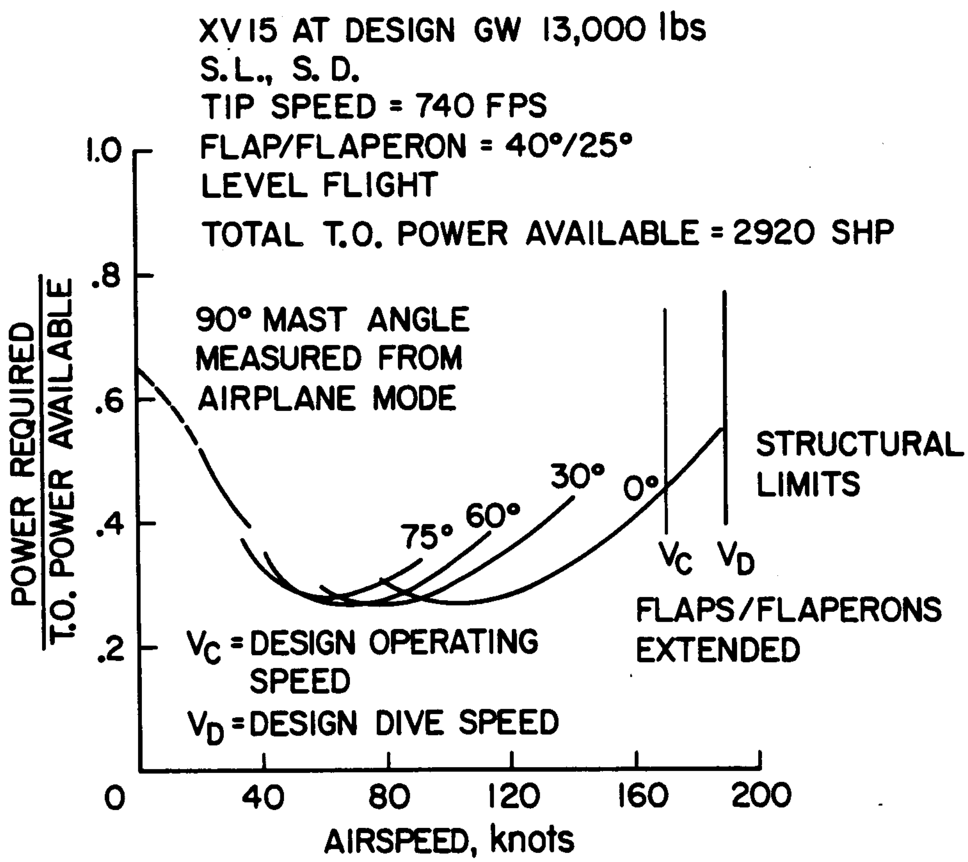


Figure 5.4.2.- Conversion power required.

5.5 Airplane Mode Flight Envelope

Figure 5.5.1 illustrates the two engine and single engine airplane mode flight envelopes at various power levels. The effect of gross weight on maximum speed is presented in figure 5.5.2.

5.6 Airplane and Helicopter Mode Endurance

The airplane mode endurance at altitudes of sea level, 10,000 ft, and 20,000 ft is shown in figure 5.6.1. Helicopter mode endurance for S.L. is also included in that figure.

5.7 Vertical Velocity Performance

Figure 5.7.1 shows the low speed climb and descent vertical velocity capabilities of the XV-15 with the flaps and flaperons extended. Both the helicopter flight mode and the airplane mode (mast angle = 0°) are examined in the climb rate.

5.7.1 *Climb performance* - The rate of climb capability of the Tilt Rotor Research Aircraft in the helicopter mode (at 80 knots) is presented in figure 5.7.1.1. Figure 5.7.1.2 depicts the maximum rate of climb for airplane mode flight.

5.7.2 *Autorotation performance* - A sink rate of less than 2400 ft/min (40 ft/s) is predicted for the steady state, autorotative condition. A flare manuever will be used to reduce this rate-of-sink just prior to touchdown.

5.8 STOL Performance

An example of the STOL capability of the Tilt Rotor Research Aircraft is indicated in figure 5.8.1. Minimum STOL take-off distance at 5,000-ft altitude, 95° F is estimated to be about 400 ft at the design gross weight.

5.9 Noise

5.9.1 *External* - Estimates of noise footprints for steep departure and takeoff angles (to a VTOL takeoff and landing) are shown on figure 5.9.1.1. The contours are superimposed on a sketch of the Moffett Field vicinity, with liftoff or touchdown at midrunway. No adverse effects on populated areas are expected.

The noise time history for an observer under the flight path one nautical mile from the touchdown or takeoff point is presented in figure 5.9.1.2. Also depicted in this figure is the time history of a research aircraft flyby at 200 knots and 1000-ft altitude. The peak noise level for this condition is less than 65 PNdB.

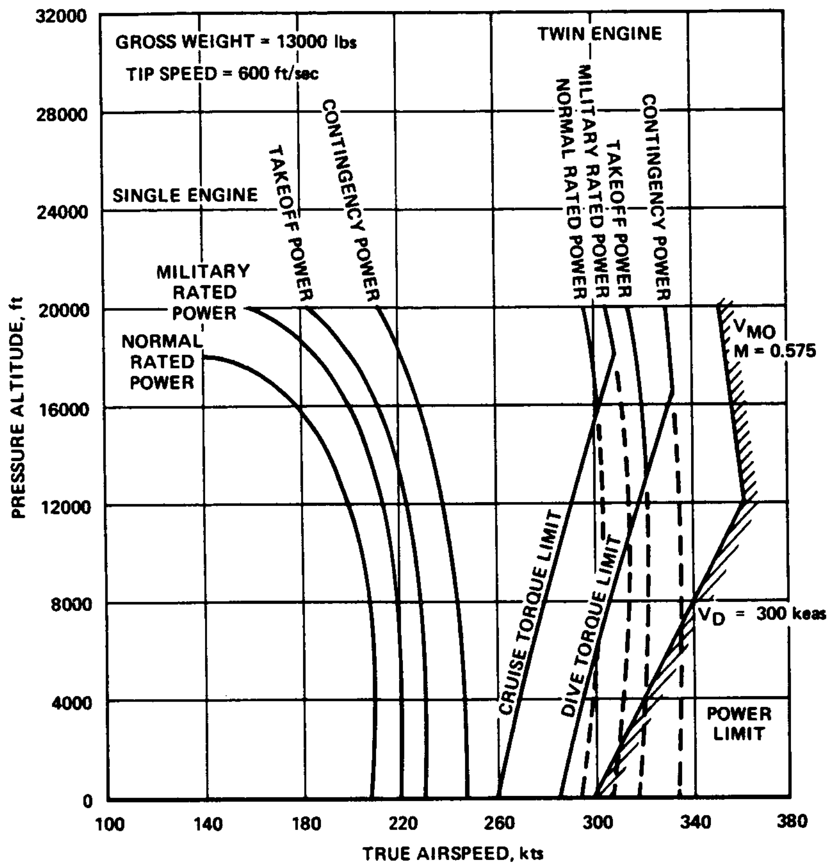


Figure 5.5.1.- Airplane mode flight envelope, standard day.

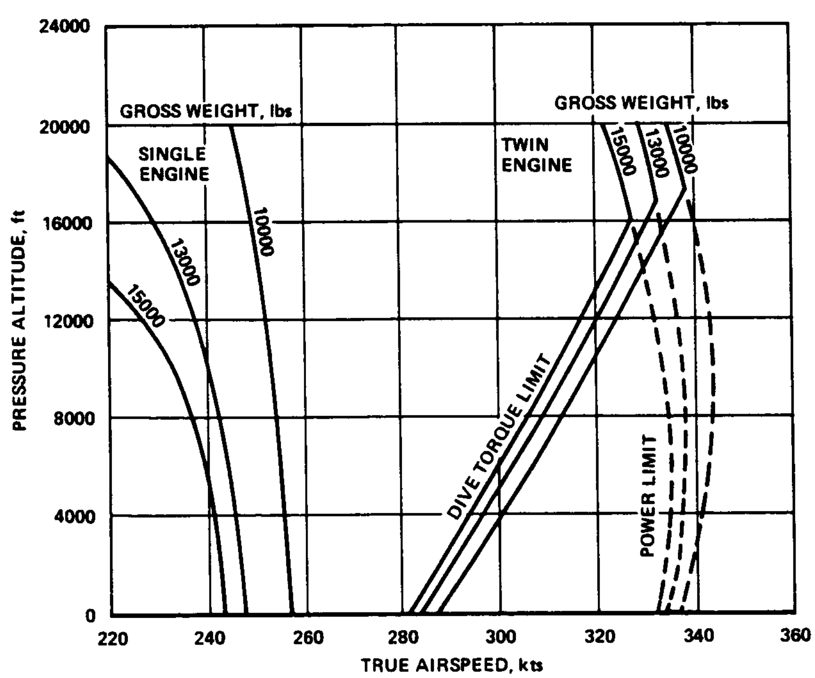


Figure 5.5.2.- Airplane mode maximum speed, contingency power, standard day.

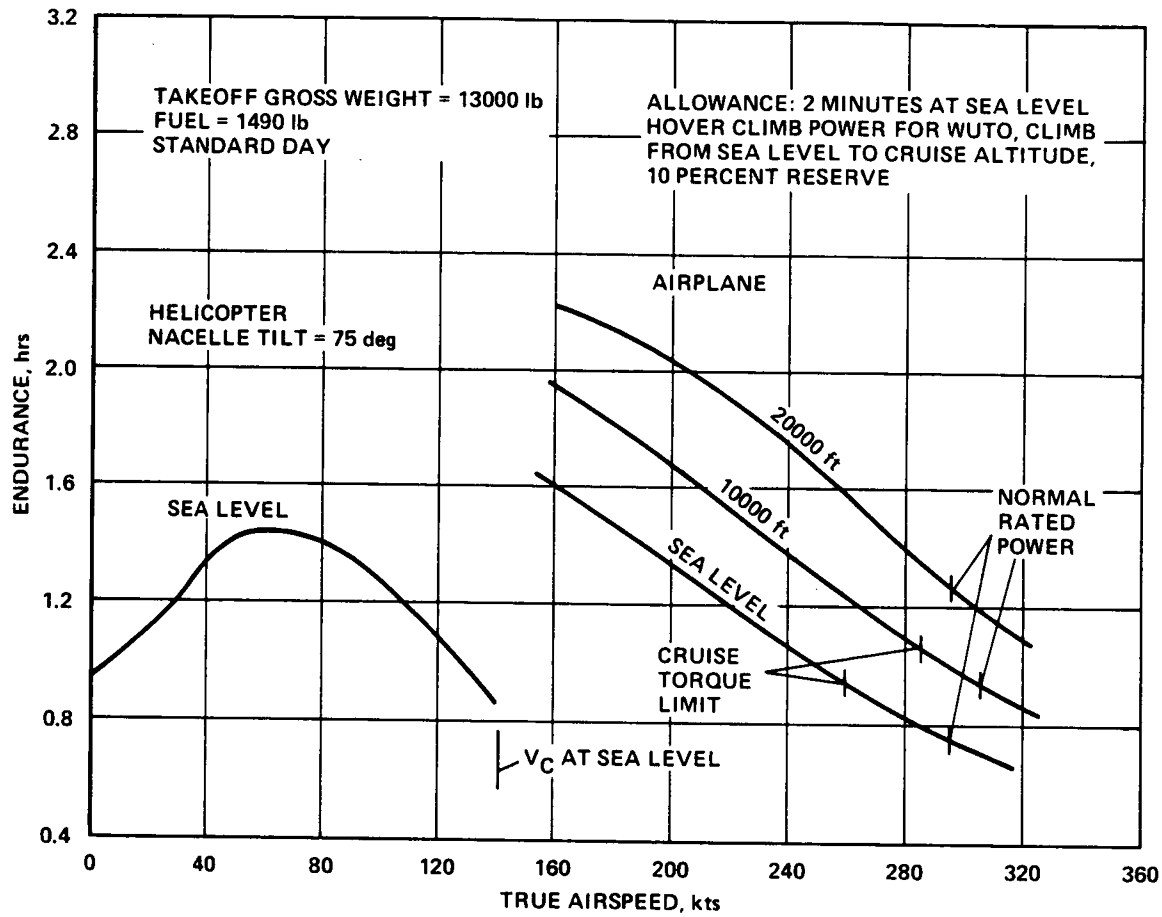


Figure 5.6.1.- Variation of endurance with airspeed.

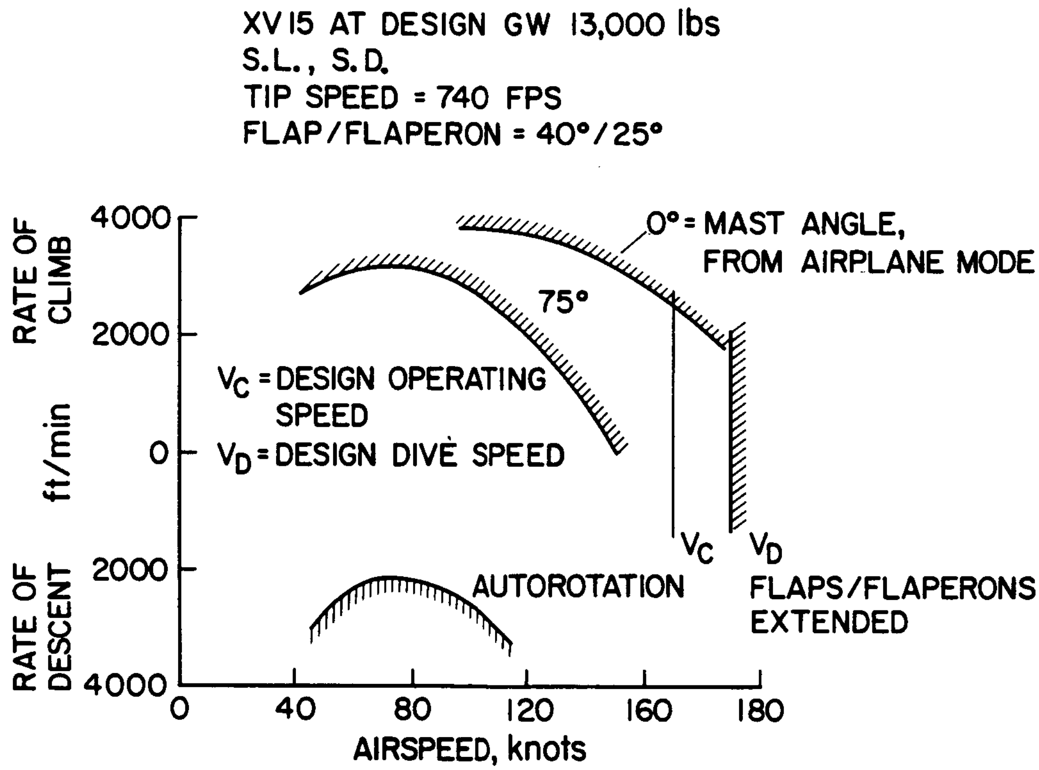


Figure 5.7.1.- Vertical velocity performance.

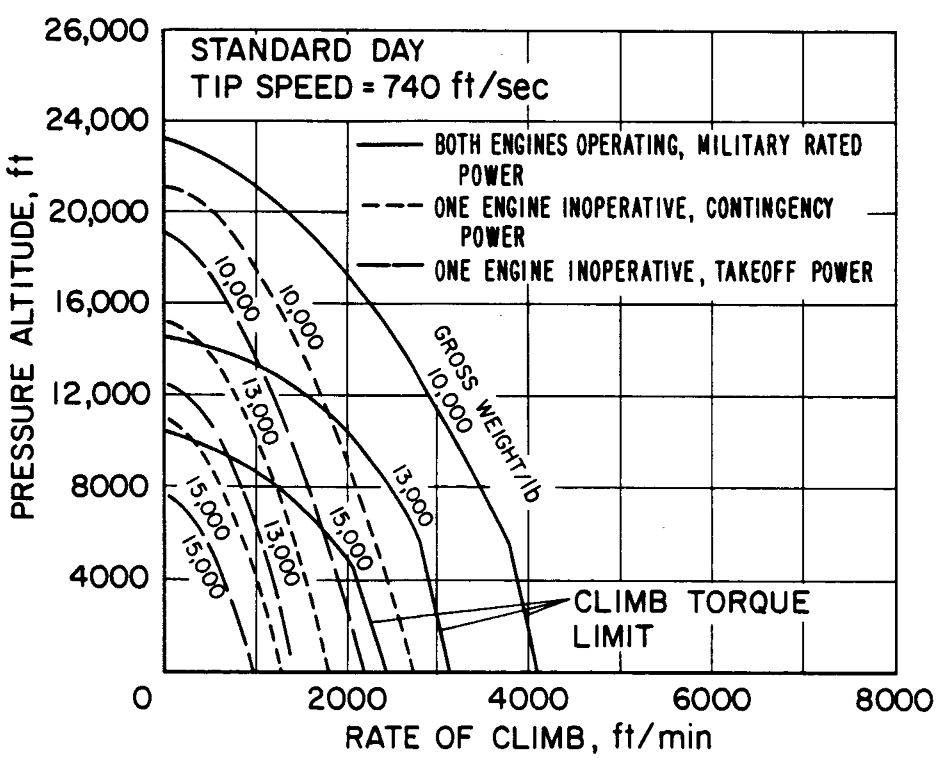


Figure 5.7.1.1.- Helicopter mode maximum rate of climb.

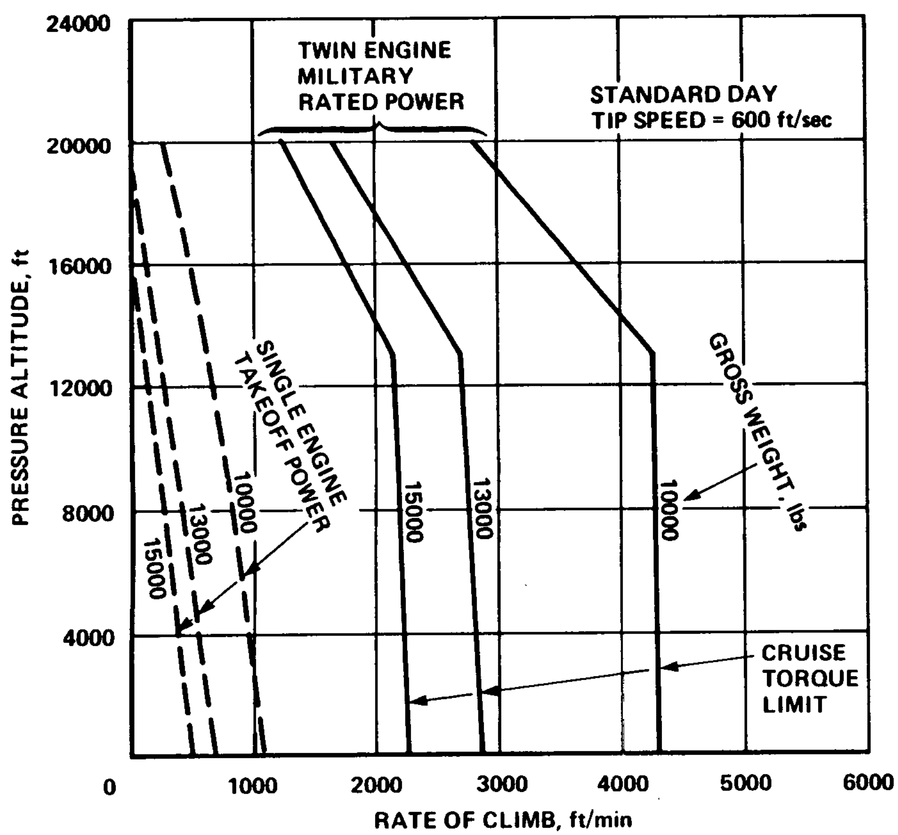


Figure 5.7.1.2.- Airplane mode maximum rate of climb.

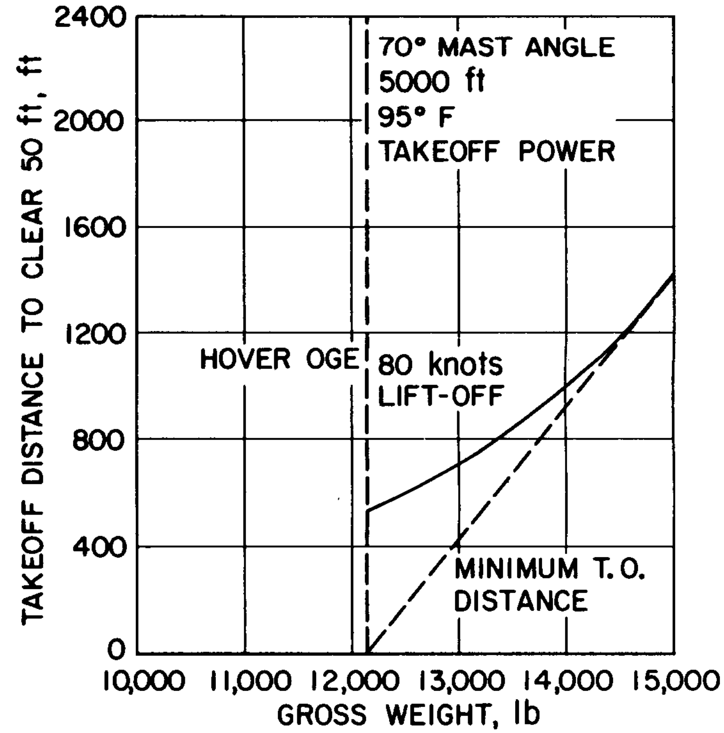
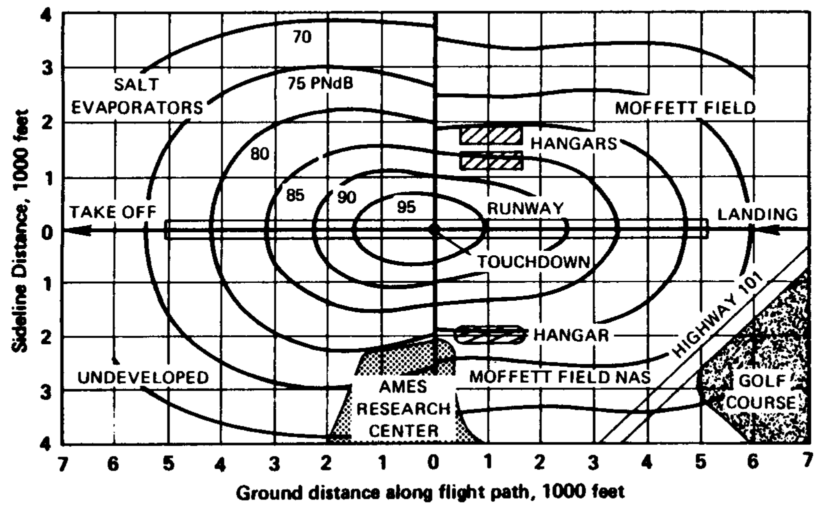
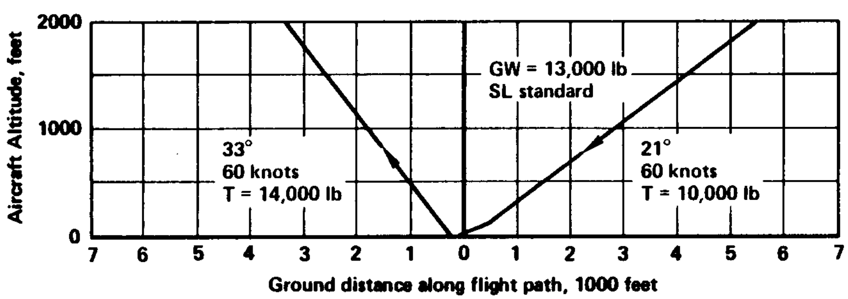


Figure 5.8.1.- STOL takeoff distance.



(a) Noise exposure contour, Moffett Field vicinity.



(b) Takeoff and landing profile.

Figure 5.9.1.1.- Noise contours.

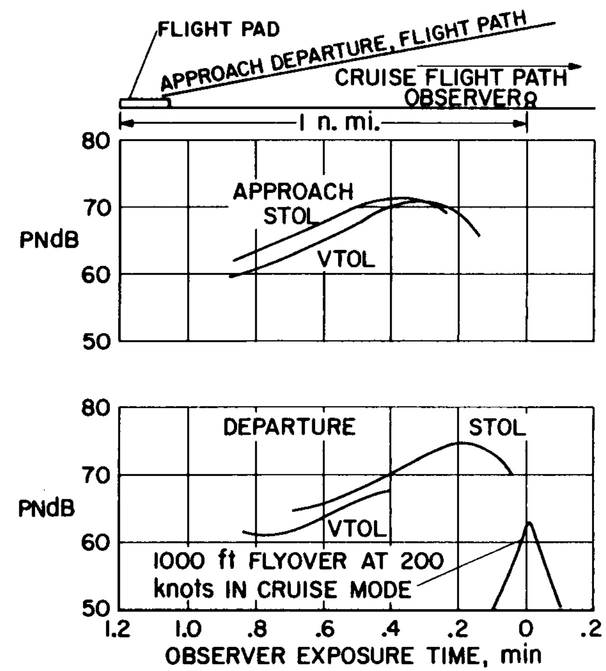


Figure 5.9.1.2.- Noise time histories.

5.9.2 *Internal* - The noise level at the crew station due to the rotors and engines will remain below desirable levels established in MIL-A-8806A throughout the significant frequency ranges. No sound-suppressant material is used to attenuate the externally generated noise. Acoustic treatment of the environmental control ducting will be used to reduce the potential environmental control unit turbine and compressor noise problem. Center-gearbox-generated noise, another potential problem, was still under study at the time of this writing. Additional protection from internal noise will be obtained through the use of the standard flight helmet.

6. ROTOR/PROPULSION SYSTEM

The system described here consists of the rotor assemblies, transmissions, cross-shafting, and powerplant. Rotor controls are described in section 8.

The Tilt Rotor Research Aircraft propulsion system is illustrated in figure 6.1. The free turbine engines drive the rotors through the main transmission located within the tilting nacelles. A cross-shafting system directly links the two main transmissions and includes a center gearbox which accommodates the angular intersection of the left- and right-hand interconnect drive shafts.

6.1 Transmissions and Cross-Shafting

Power is transmitted from the engine to the rotor through a coupling gearbox and the main transmission. The main structural components of this assembly are the steel spindle, which serves as the axis about which the nacelle rotates, and the transmission castings. The engine is cantilevered from the coupling gearbox casting. The nacelle cowlings are supported by the transmission case.

The coupling gearbox contains three herringbone gears which increase rotational speed from an engine rpm of approximately 20,000 to about 30,000 rpm at the main transmission interface. The main transmission consists of the primary reduction stages after the coupling gearbox output, and the two-stage planetary reduction gears that drive the rotor shaft at 565 rpm in hover. High helix angle, fine-pitch heringbone gears in the high-speed train reduce friction loss and noise. The planetary systems are also low friction, low noise, high contact ratio, relatively fine-pitch gears. Coulomb (friction) damping devices on the herringbone gear rims and webs preclude large magnitude "cymbal" resonant modes. The interconnect drive system consists of a spur gear set and a bevel gear set in each main transmission. Both rotors are always directly connected through the interconnect drive train and the planetary gears. Accessory drive trains and accessory pads on each main transmission casing are provided. A one-way clutch is provided between the high-speed reduction stage and the planetary reduction units to disengage the engine in the event of a power failure. The interconnect drive shaft system is linked to the rotor side of the one-way clutch so that power to both rotors is available with either engine shut down.

A schematic illustration of the drive system gear ratios is presented in figure 6.1.1.

The transmission is designed to meet a 3,000-hour overhaul life.

6.2 Engine

Two modified Lycoming T53-L13B turboshaft engines power the Tilt Rotor Research Aircraft. The engines are modified as follows:

- Nose gearbox removed and direct drive provisions added,
- Vertical starting, operating, and stowing capabilities added (same as made for LTCK-4C engines in Canadair CL-84),
- Replacement of the last stage power turbine disk with that from the Lycoming T5319A engine to provide overspeed capability, and
- Replacement of the first stage gas producer turbine blades with those from the Lycoming T5319A engine to provide the two minute contingency rating.

The key configuration and performance characteristics are described in the following table and accompanying figures:

Manufacturer and designation:	Lycoming LTC1K-41K
Installation:	In tilting nacelle (fig. 6.2.1)
Shaft rotation:	Clockwise (viewed from rear)

Rating (sea level standard day):

	shp	sfc
Contingency (2 minutes)	1802	0.564
Takeoff (10 minutes)	1550	.584
Military (30 minutes)	1401	.601
Normal (max. continuous)	1250	.622

Power available, fuel flow, and jet thrust in helicopter, conversion and airplane mode are shown in figures 6.2.2 to 6.2.14.

A listing of the exhaust emission levels expected for the LTC1K-4K is presented below.

Exhaust emissions from both T-53-L13A engines (typical):

	Ground idle at 57 shp (lb/h)	30-min rating at 1400 shp (lb/h)
Carbon monoxide	7.7	1.2
Carbon dioxide	2.4	4.0
Hydrocarbons	2.7	0.06
Oxides of nitrogen	0.2	4.6

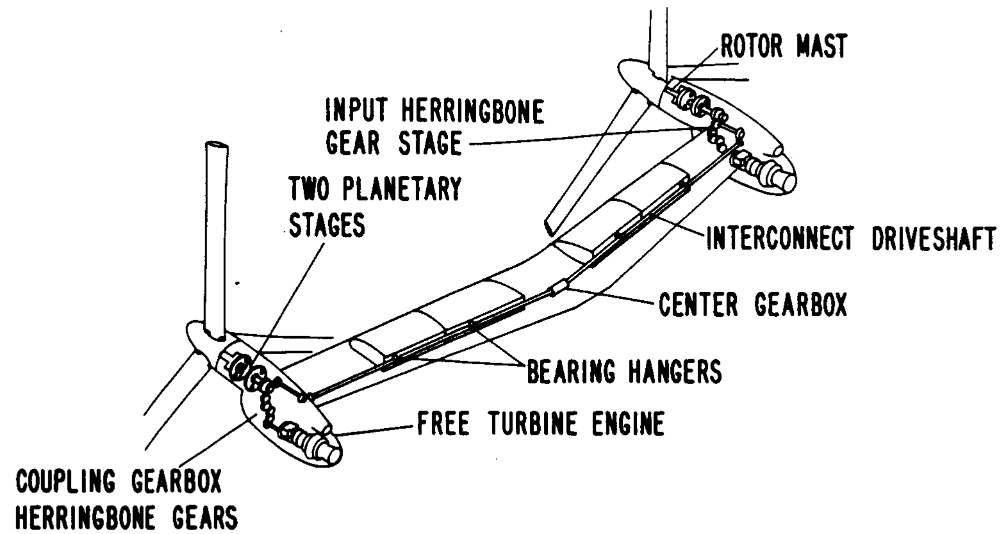


Figure 6.1.- Propulsion system.

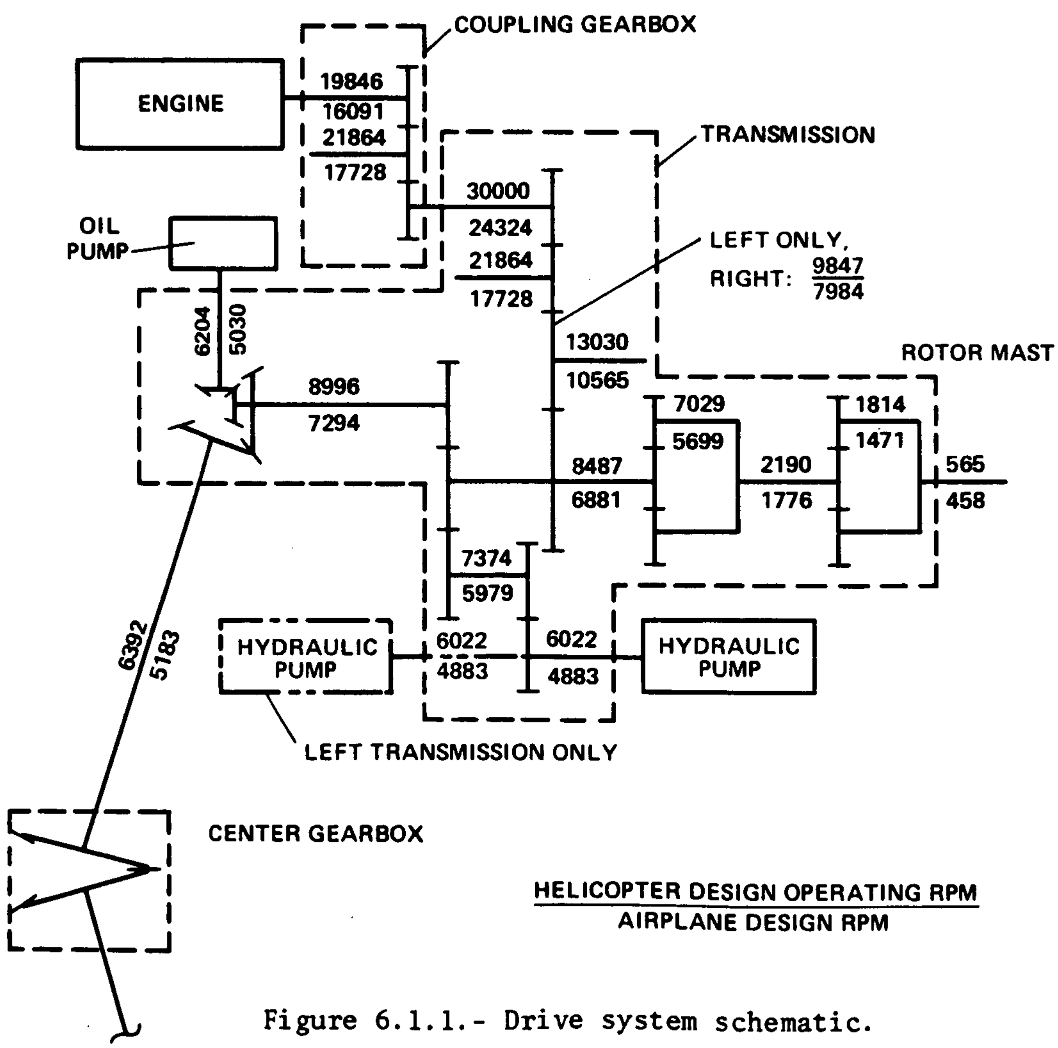


Figure 6.1.1.- Drive system schematic.

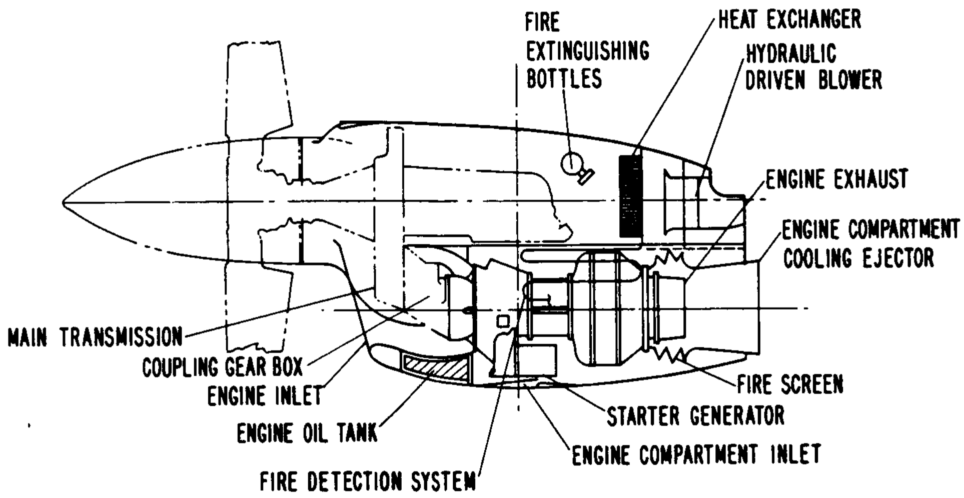


Figure 6.2.1.- Engine installation

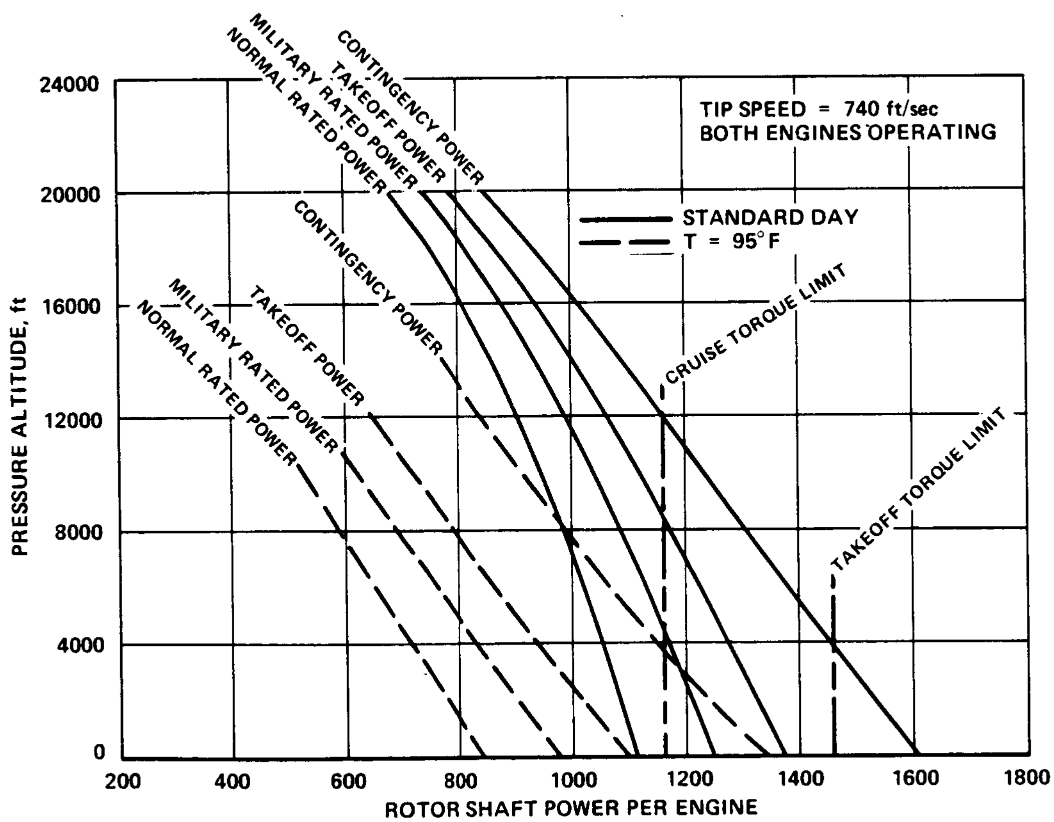


Figure 6.2.2.- Power available, helicopter and conversion.

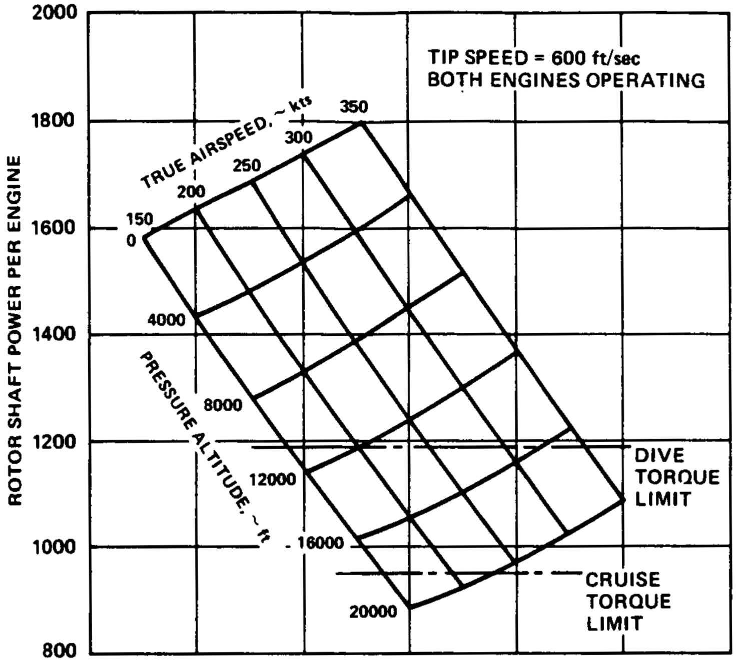


Figure 6.2.3.- Contingency power available, airplane mode, standard day.

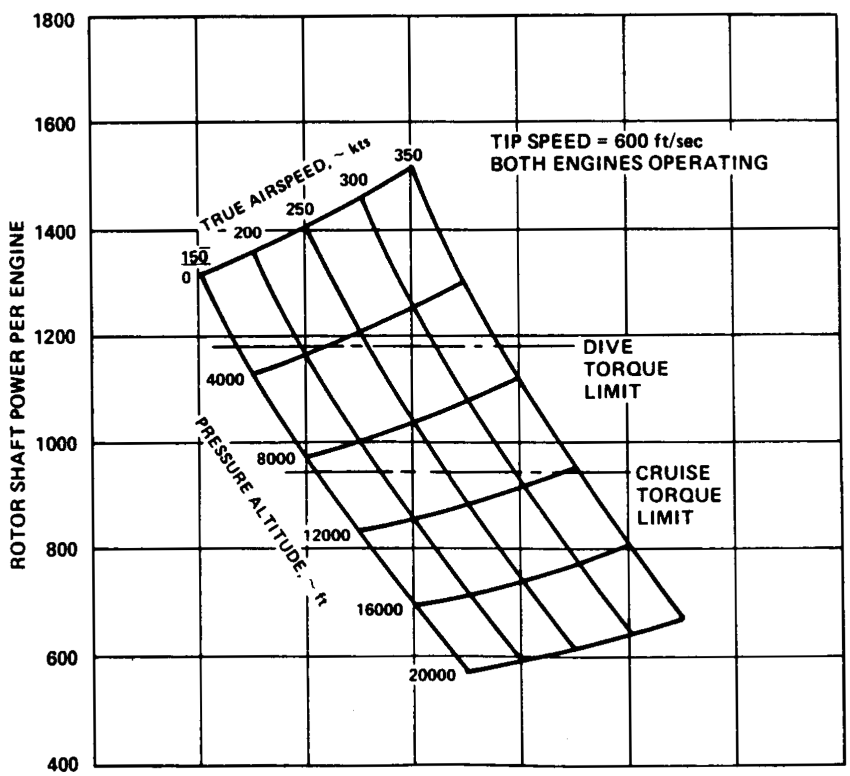


Figure 6.2.4.- Contingency power available, airplane mode, T = 95° F.

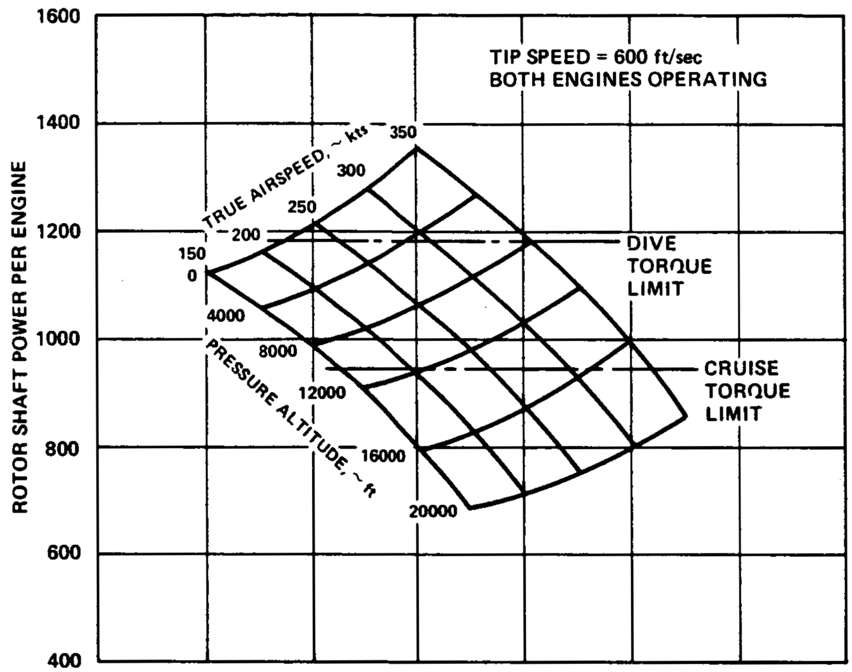


Figure 6.2.5.- Normal rated power available, airplane mode, standard day.

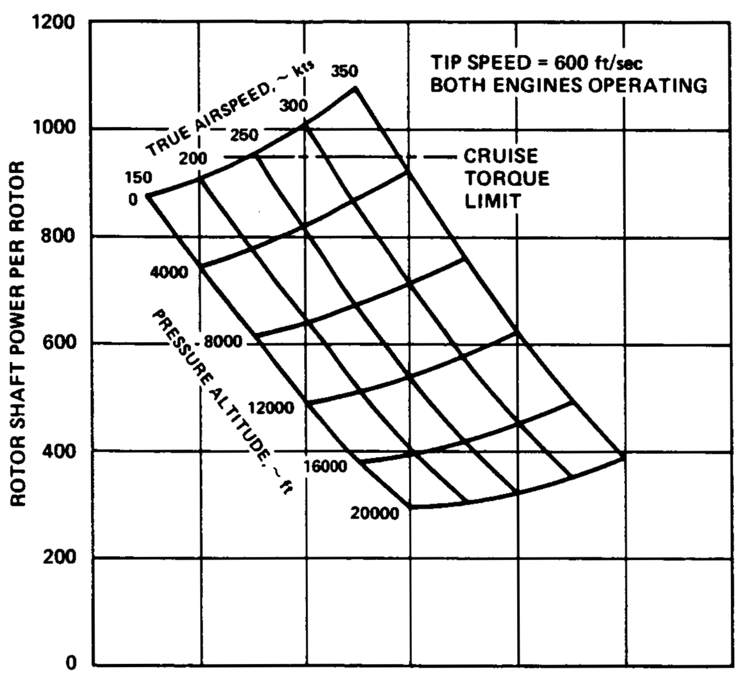


Figure 6.2.6.- Normal rated power available, airplane mode, T = 95° F.

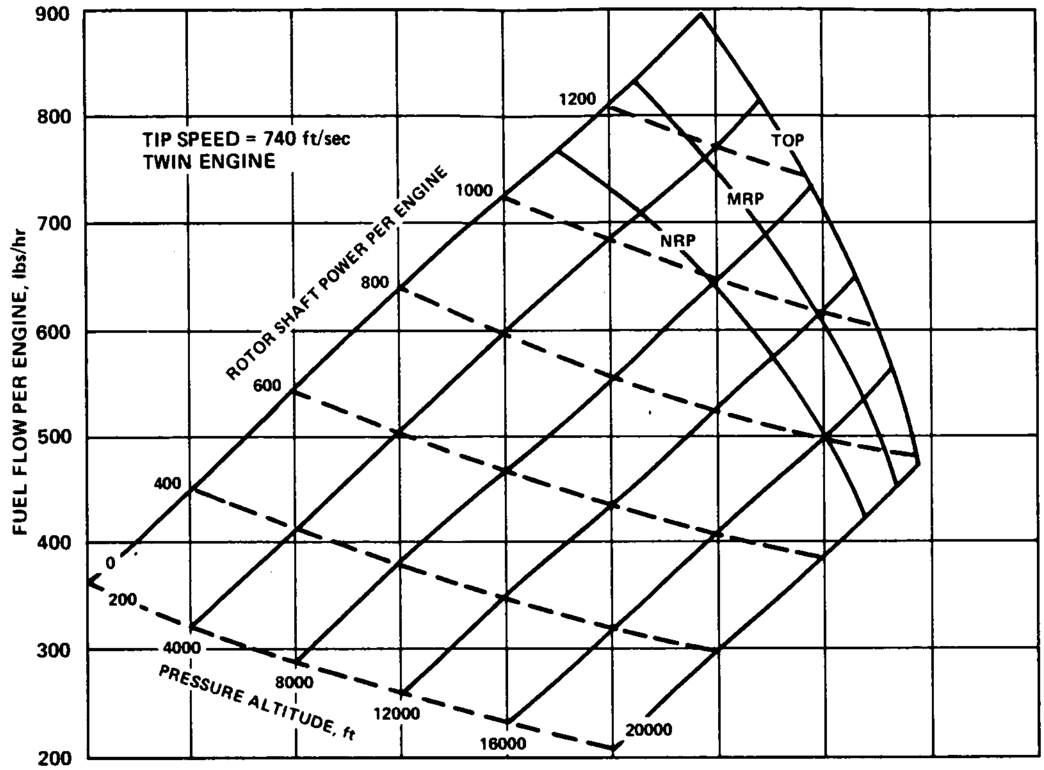


Figure 6.2.7.- Fuel flow, helicopter and conversion, standard day.

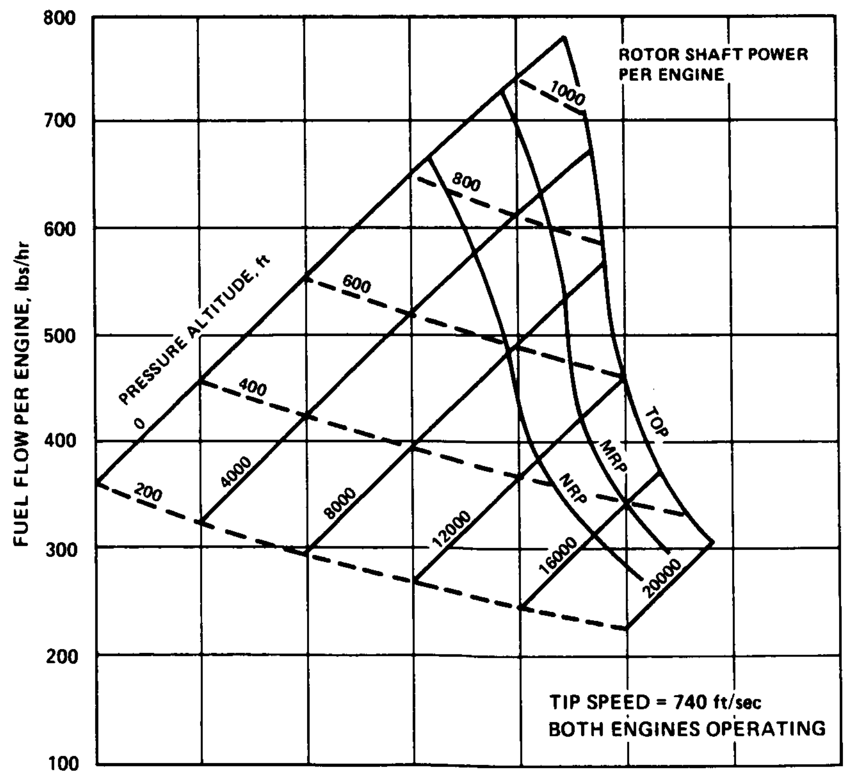


Figure 6.2.8.- Fuel flow, helicopter and conversion, T = 95° F.

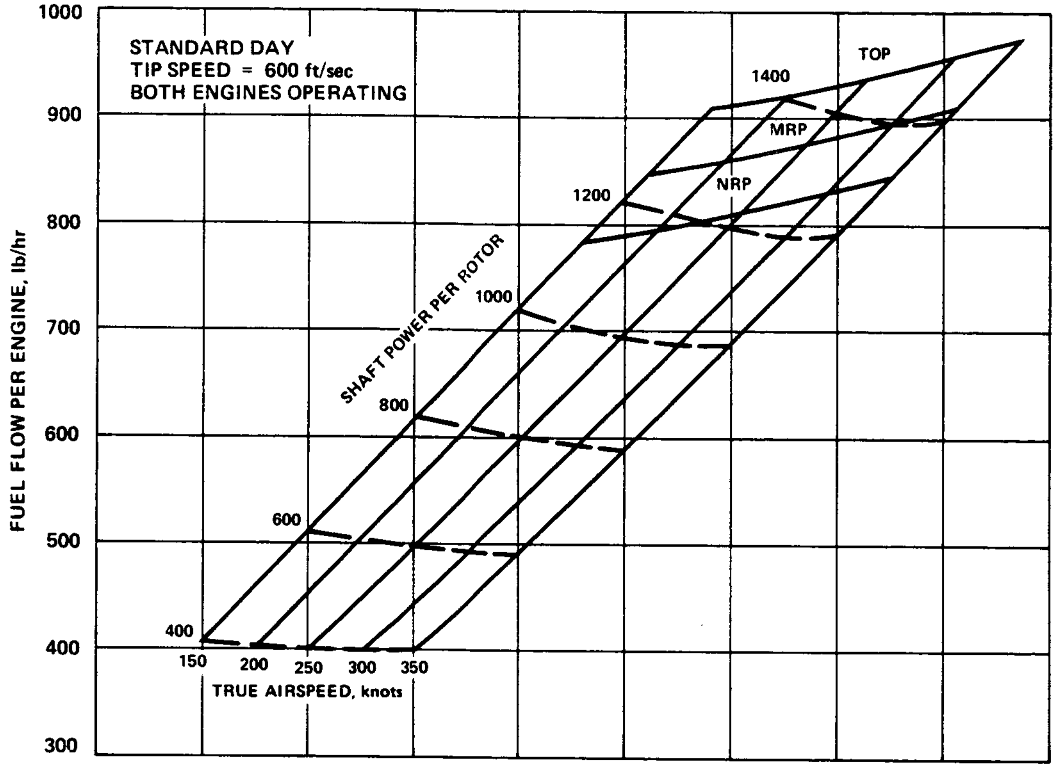


Figure 6.2.9.- Fuel flow, airplane mode, sea level.

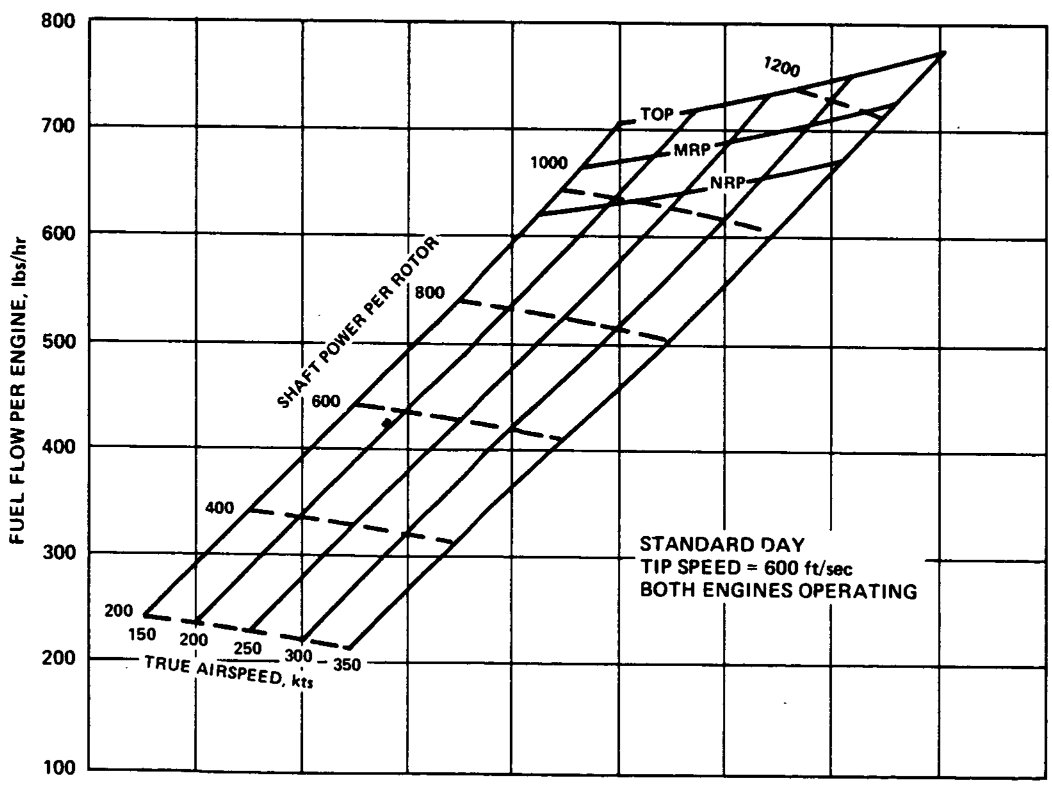


Figure 6.2.10.- Fuel flow, airplane mode, 10,000 ft altitude.

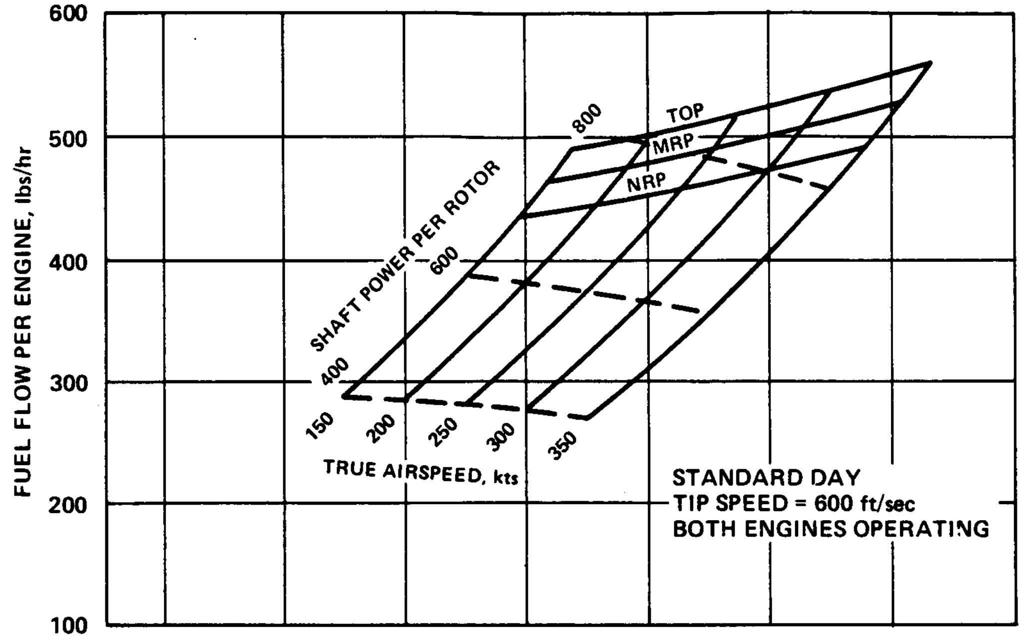


Figure 6.2.11.- Fuel flow, airplane mode, 20,000 ft altitude.

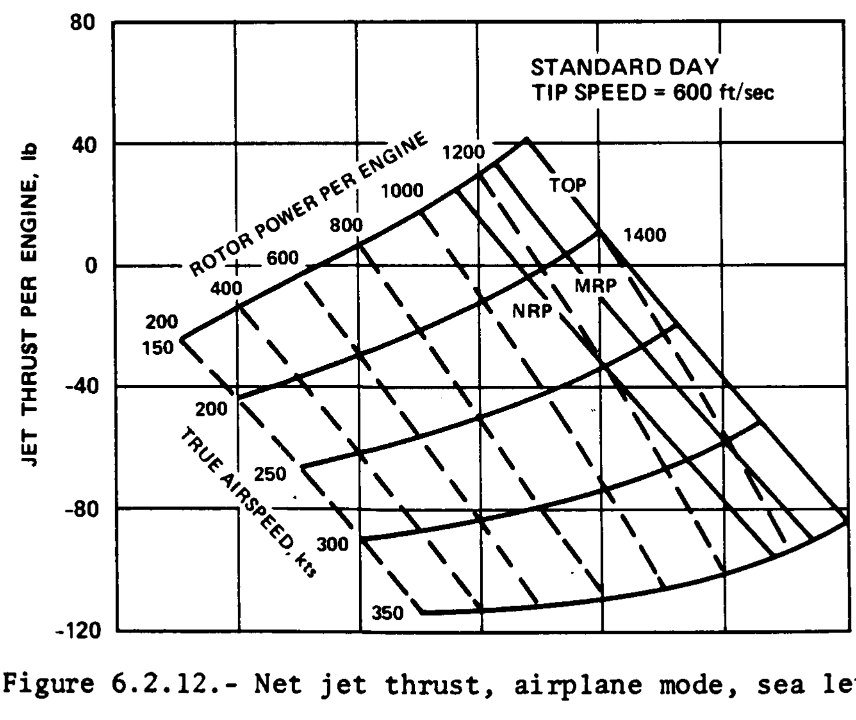


Figure 6.2.12.- Net jet thrust, airplane mode, sea level.

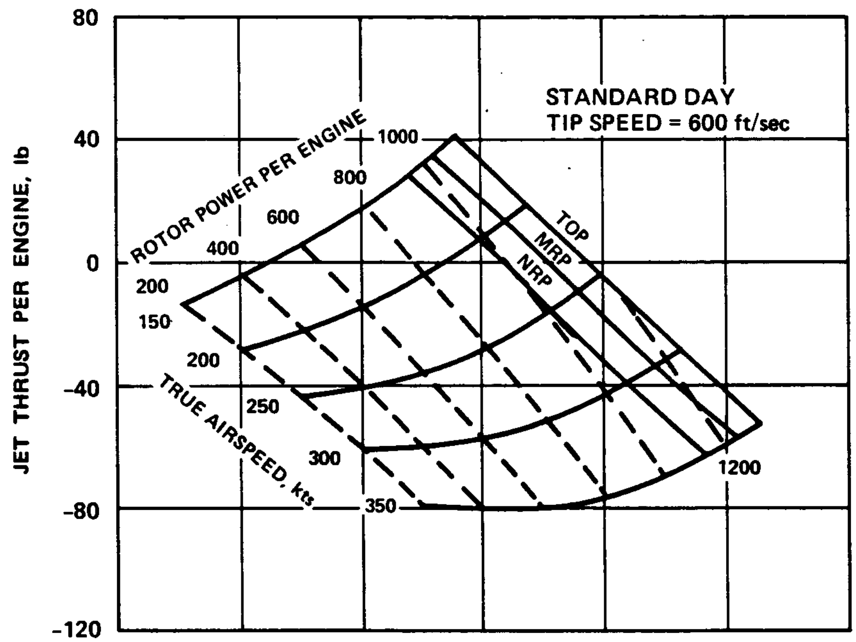


Figure 6.2.13.- Net jet thrust, airplane mode, 10,000 ft altitude.

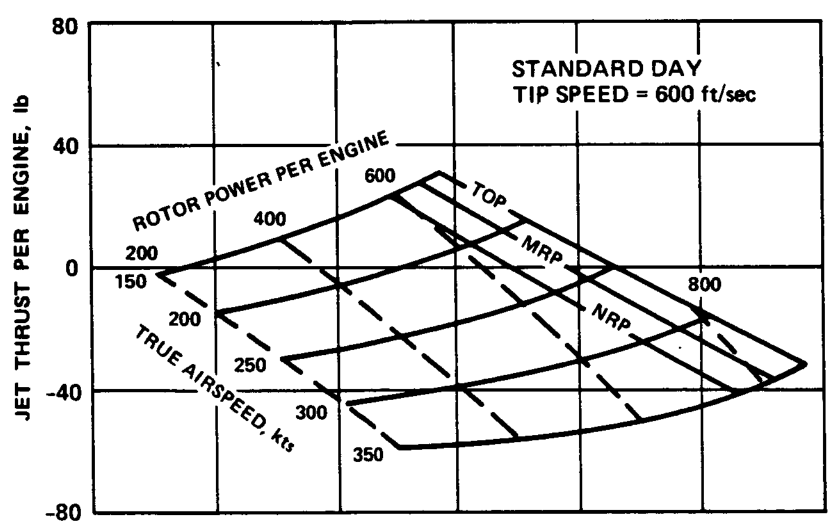


Figure 6.2.14.- Net jet thrust, airplane mode, 20,000 ft altitude.

6.3 Rotor

The XV-15 has two 25 ft diameter three-bladed rotors. Wire-wound tension-torsion straps retain each of the blades to a titanium yoke which is gimbal mounted to the mast (fig. 6.3.1). A nonrotating rubber hub-moment spring (fig. 6.3.2) provides improved longitudinal control power. The 14-inch chord blades are constructed of 17-7 PH stainless steel and are NACA 64 series airfoils. They are 8 percent thick at the tip and 35 percent thick at the theoretical root. The blades are twisted to provide high performance levels both in the hover and in high speed cruise (airplane) modes. A typical blade section is shown in figure 6.3.3. The components are bonded together in a blade cavity tool to provide close contour control. The intercised aluminum honeycomb core in the aft portion of the blades provides contour stabilization. The blades are mass balanced by the nose weight. The rotor has cyclic and collective control through a pitch horn which is placed to provide positive pitch-flap coupling (unconventional delta-three) for improved stability.

7. STRUCTURAL DYNAMICS

7.1 Coupled Natural Frequencies

Rotor - The coupled natural frequencies of the rotor are shown in figures 7.1.1 and 7.1.2 in terms of collective modes and cyclic modes, respectively. The collective modes are those having polar symmetry about the mast. The cyclic modes are those asymmetric with respect to the mast. A major feature of the XV-15 rotor system is the placement of the lowest inplane natural frequency above the rotor operating speed (1/rev). This eliminates the possibility of ground or air resonance as has been demonstrated with previous stiff-inplane designs. Three modes (third collective mode and second and third cyclic modes) that are indicated to be near resonant conditions at certain portions of the helicopter or airplane flight envelope were monitored during the wind tunnel test. These modes were found to be satisfactorily placed on the basis of the tunnel tests.

Drive System - The drive system undamped natural frequencies and normalized mode shapes are shown in figure 7.1.3 for the symmetrical oscillatory rotation (no differential torque between left- and right-hand systems) and the asymmetric rotation modes (i.e., right- and left-hand relative torsional deflection in the same rotation direction).

The principal oscillatory frequency of aerodynamic excitation is three-per-rev. The rotor's flapping gimbal causes two-per-rev torques. These bands of excitation are noted in the figure. The drive system natural frequencies are seen to be well located with regard to excitation bands.

Interconnect shaft frequencies are placed to prevent shaft whirl instability and are well removed from rotor three-per-rev excitation.

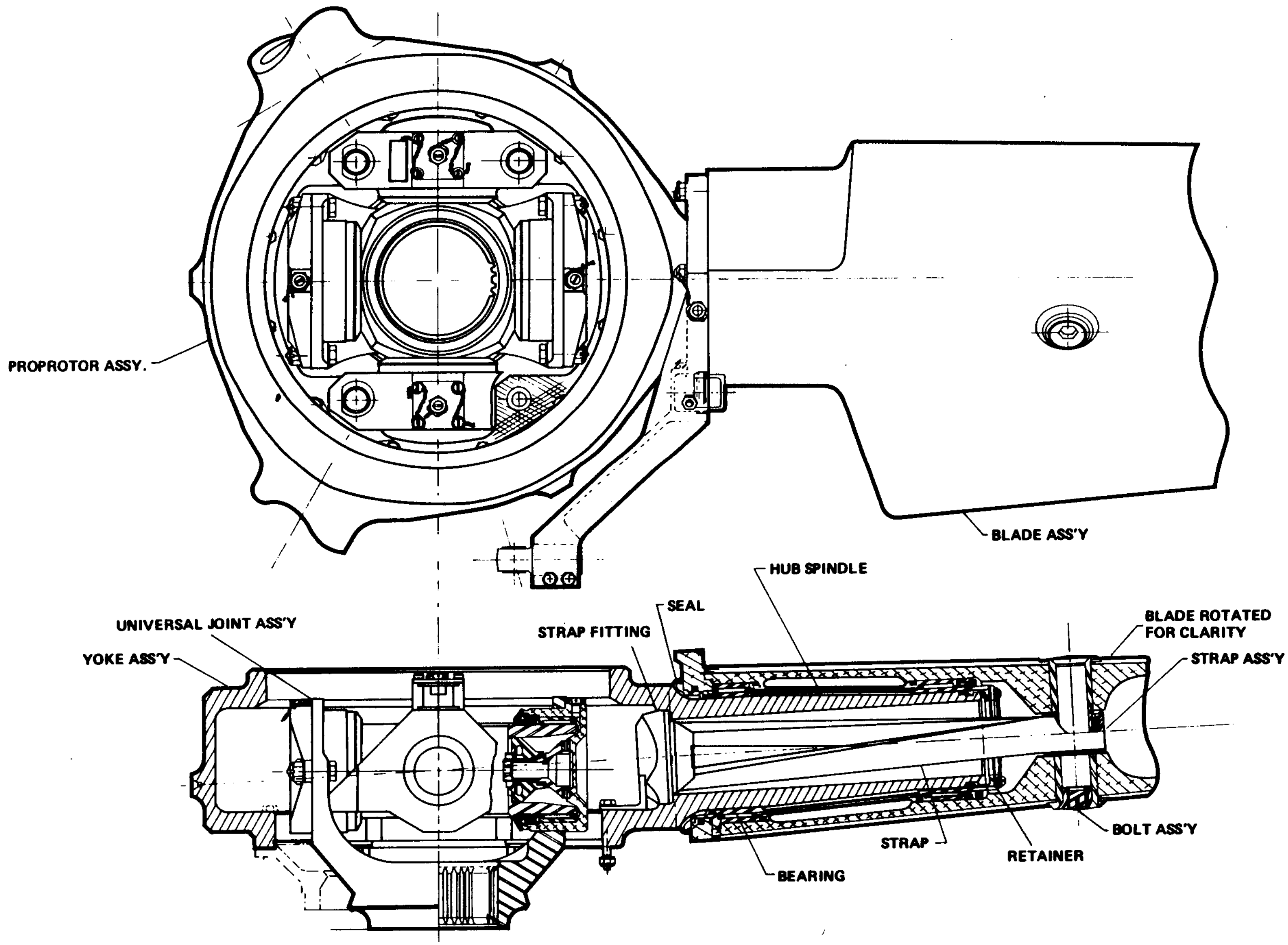


Figure 6.3.1.- Hub and blade retention assemblies.

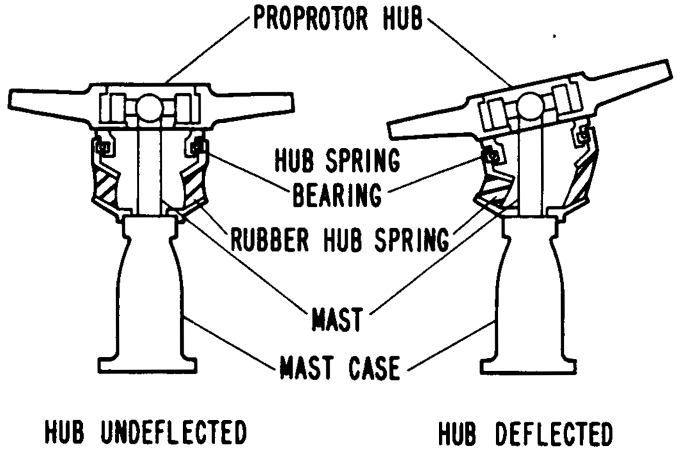


Figure 6.3.2.- Hub spring schematic.

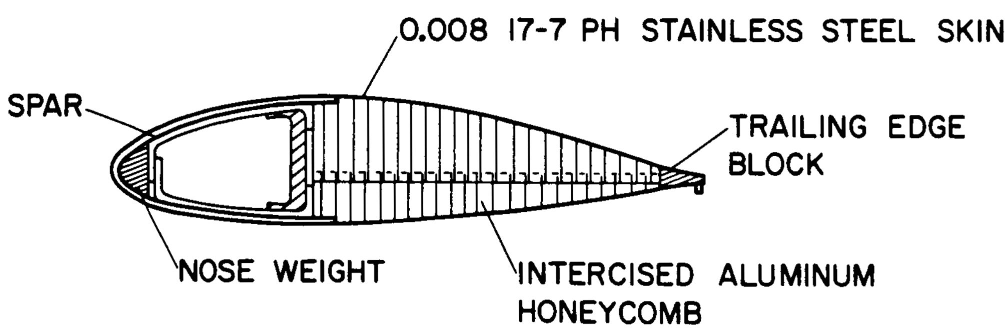


Figure 6.3.3.- Typical blade section.

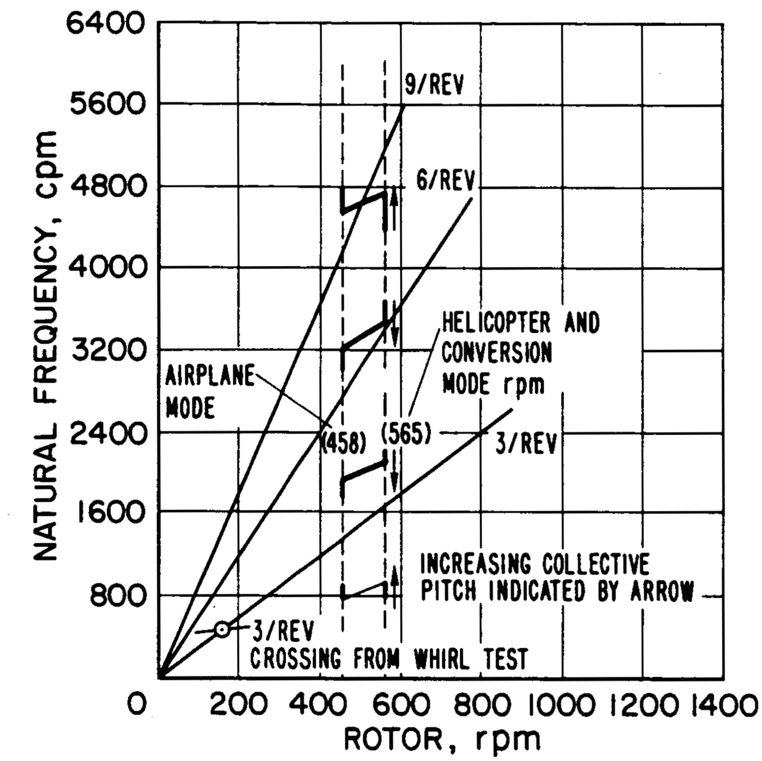


Figure 7.1.1.- Rotor collective modes.

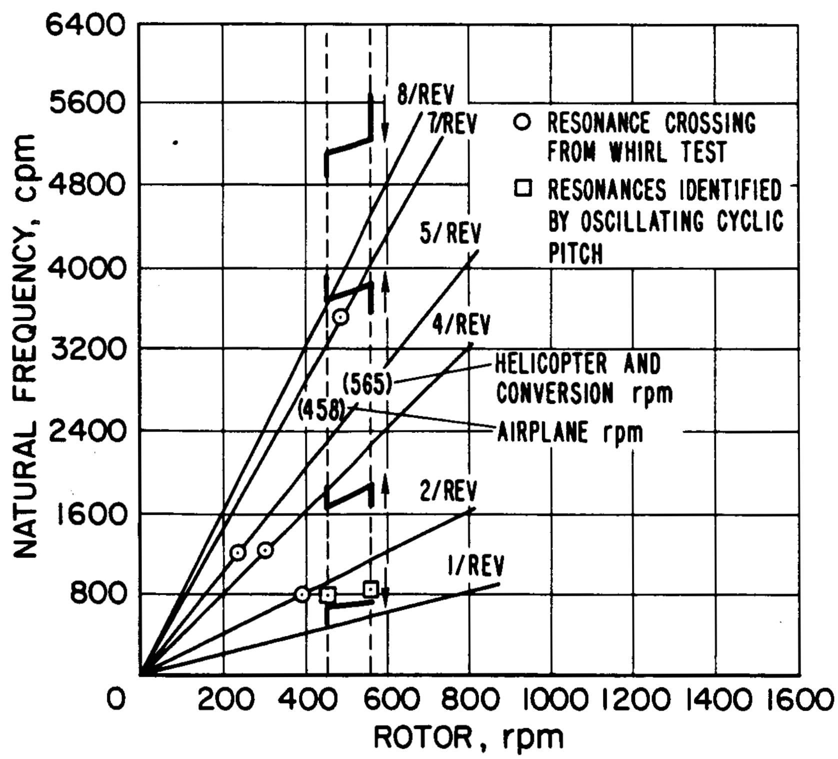
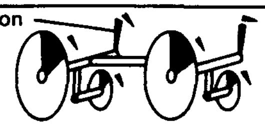
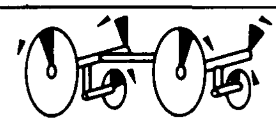
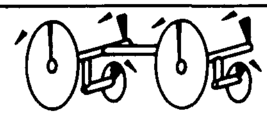
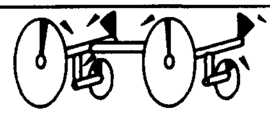
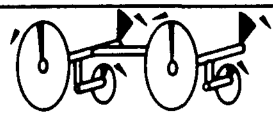


Figure 7.1.2.- Rotor cyclic modes.

Torsional modes	Frequency (per rev)		Mode shapes
	565 rpm	458 rpm	
1st asymmetric flexible mode (interconnect shaft mode)	.39	.48	
1st symmetric flexible mode (symmetric power turbine mode)	1.26	1.55	
2nd asymmetric flexible mode (asymmetric power turbine mode)	1.43	1.76	
2nd symmetric flexible mode (symmetric pylon roll mode)	3.42	4.22	
3rd asymmetric mode (asymmetric pylon roll)	3.49	4.30	

Note: Engine rotations referred to tilt rotor
Positive engine rotation same as corresponding tilt rotor

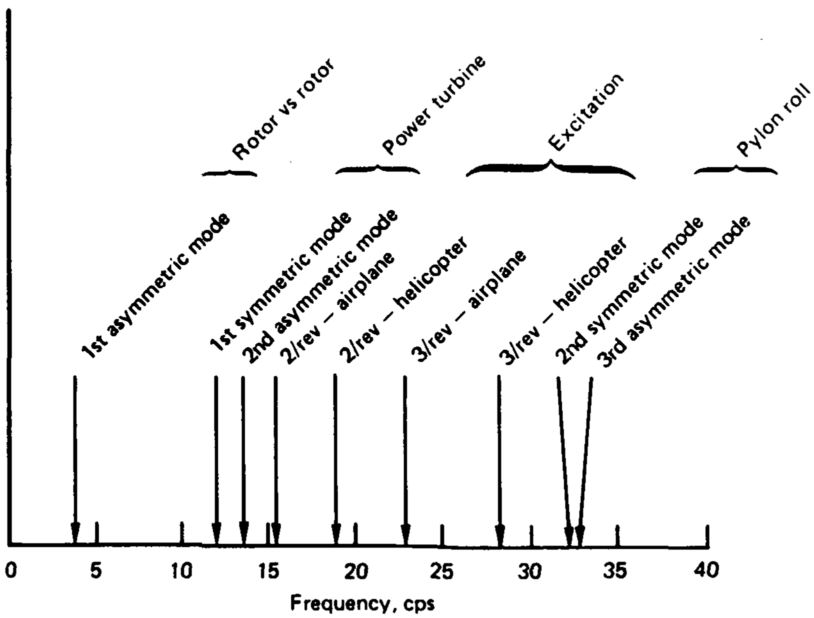


Figure 7.1.3.- Drive system natural frequencies.

Airframe - The calculated natural frequencies of the wing-pylon-fuselage-empennage system as a function of pylon conversion angle are shown in figure 7.1.4 (for both the symmetric and asymmetric modes). The natural frequency variation with pylon conversion angle is a result in the shift of the pylon center of mass as the pylon is converted. The sudden frequency change at zero degrees is caused by engagement of the pylon downstops. The wing torsion modes are calculated to be near or in resonance with 1/rev at airplane mode rpm. If, during the ground shake tests of the aircraft, the actual wing torsion frequency turns out to be exactly as predicted, one of two simple alternatives will be selected. The airplane mode rpm can be changed slightly to avoid the resonance with only minor performance changes, or the aluminum downstops can be replaced with steel to increase the mode frequency. Other modes that are associated with the fuselage and empennage have been calculated based on preliminary mass and stiffness data. These modes will be reanalyzed as actual design properties become available and stiffnesses will be changed as necessary to avoid resonances.

7.2 Aeroelastic Stability

The stability boundaries for the lowest conversion mode stability condition, with pylons nearly in the airplane mode ($\theta_m = 1°$) and the downstops (which increase the stability at high speeds) not yet engaged, are shown in figures 7.2.1 and 7.2.2. The wing chord and beam mode stability boundaries are shown against rotor rpm, and altitude. The conversion mode operating regions are well within the stable areas.

The calculated airplane mode stability boundaries are presented in figures 7.2.3 and 7.2.4. The most critical mode, the wing chord symmetric mode is stable through 400 knots at 458 rotor rpm. This is in excess of the flutter-free requirement of 1.2 V_{DIVE} (360 knots at sea level). Figure 7.2.4 also shows that drag divergence takes place prior to the onset of wing chord mode instability.

The effects of the loss of 20 percent of all wing-pylon stiffness (beam, chord and torsion) is shown in figures 7.2.5 (versus rpm) and 7.2.6 (versus altitude). With this loss of stiffness the XV-15 still maintains the 1.2 V_{DIVE} flutter-free margin. The Dutch roll mode boundary and the 1.2 M_{DIVE} line are essentially coincident above 12,000 feet. The stability boundary for this mode is not as critical as that of the aeroelastic modes because of its low frequency and because of the pilot's ability to control it.

Figure 7.2.7 shows that the predicted flutter boundary for the empennage occurs at airspeeds beyond the 1.2 V_{DIVE} flutter-free requirement. For the highest speed and Mach number conditions tested with the one-fifth scale model, there was no indication of flutter.

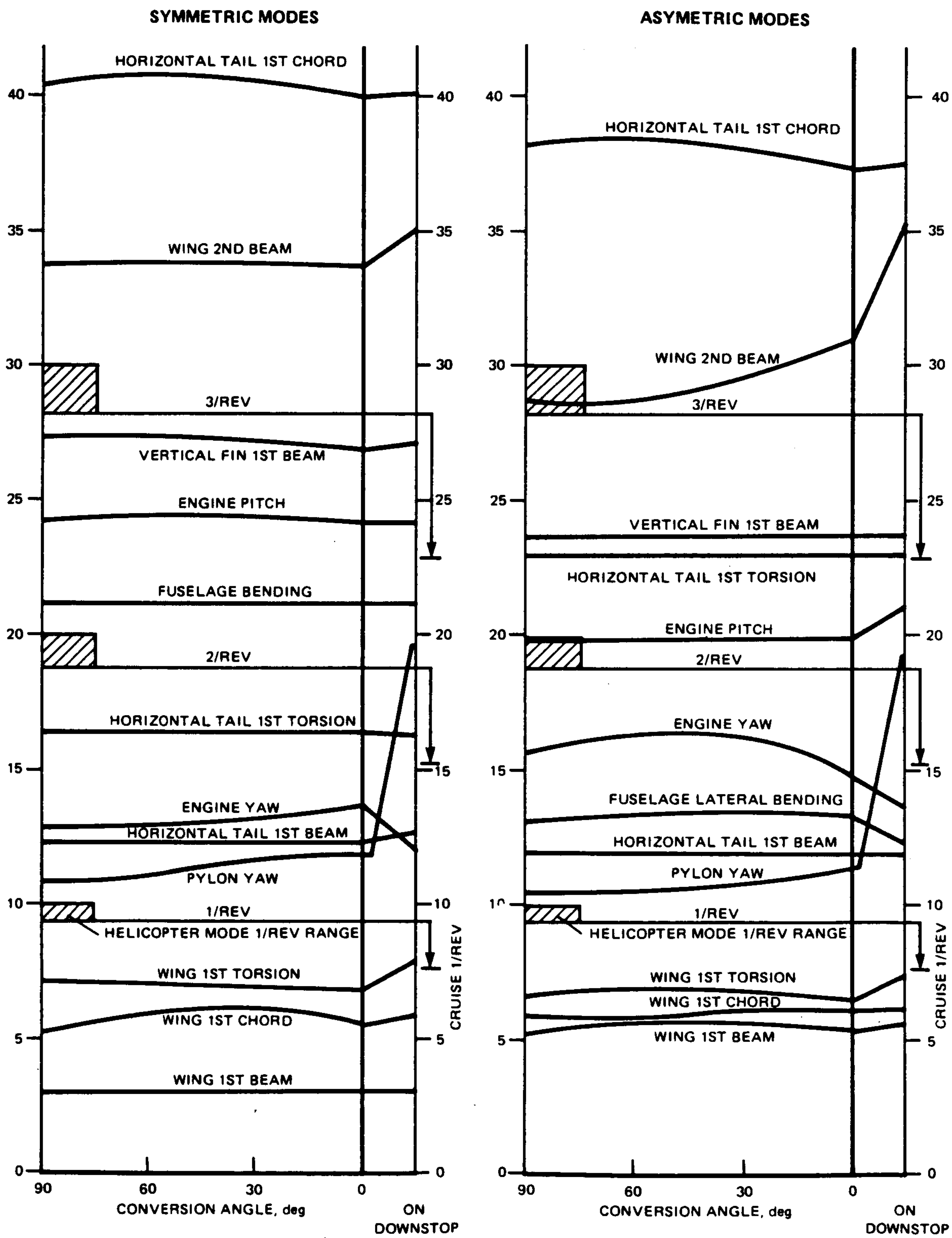


Figure 7.1.4.- Airframe natural frequencies.

7.3 Rotor Flapping

The total rotor flapping excursions for a range of flight envelope conditions, including gusts, are presented in figure 7.3.1. This figure shows that the flapping stops (at 12°) are not encountered for any of the level flight or maneuver conditions considered.

8. FLIGHT CONTROL SYSTEMS

8.1 Basic Flight Controls

The Tilt Rotor Research Aircraft control system, shown schematically in figure 8.1.1, is designed to permit single pilot operation from either seat. Control moments are generated by means of rotor and fixed surface controls, with the rotor controls phased out as the aircraft is converted from helicopter to airplane mode. The conversion and power management systems are designed for straightforward cockpit procedures. All normal and emergency procedures can be controlled by a single pilot.

8.1.1 *Cockpit controls* - The cockpit controls consist of a longitudinal/lateral stick, collective-type power lever and pedals for the pilot and copilot. The throttles are mounted on a center console which also contains levers that permit either pilot to control the flaps and landing gear and a blade-pitch-governor hand-wheel for manual override of the rotor governor. A three-position switch on each power lever controls the nacelle conversion angle.

8.1.2 *Rotor controls* - In the helicopter mode, pitching moments are generated by applying longitudinal cyclic pitch change to the rotor blades. Rolling moments are generated by applying differential collective pitch change to the rotors. (Lateral cyclic pitch is programmed as a function of nacelle incidence and airspeed to minimize rotor flapping.) Yawing moments are generated by the application of differential longitudinal cyclic pitch change to the rotors which results in opposite tilting of the rotors. Upward or downward movement of the power lever simultaneously increases or decreases engine power and rotor blade collective pitch to provide vertical thrust control.

8.1.3 *Fixed controls* - The elevator, ailerons, and rudder are active in all modes of flight. During conversion from helicopter mode to airplane mode, the desired control response is achieved by phasing out the rotor controls as the fixed controls become effective.

The wing has full-span flaps and the ailerons are actually flaperons that can be deflected differentially for roll control. The horizontal stabilizer has a 30 percent chord elevator that can be deflected ±20° for pitch control. In addition, the horizontal stabilizer incidence is ground adjustable from 20° leading-edge-up to 10° leading-edge-down to provide pitch trim.

8.1.4 *Conversion system* - This system provides controlled rotation of the nacelles over a range of 95°. The nacelles are tilted by ball-screw jack

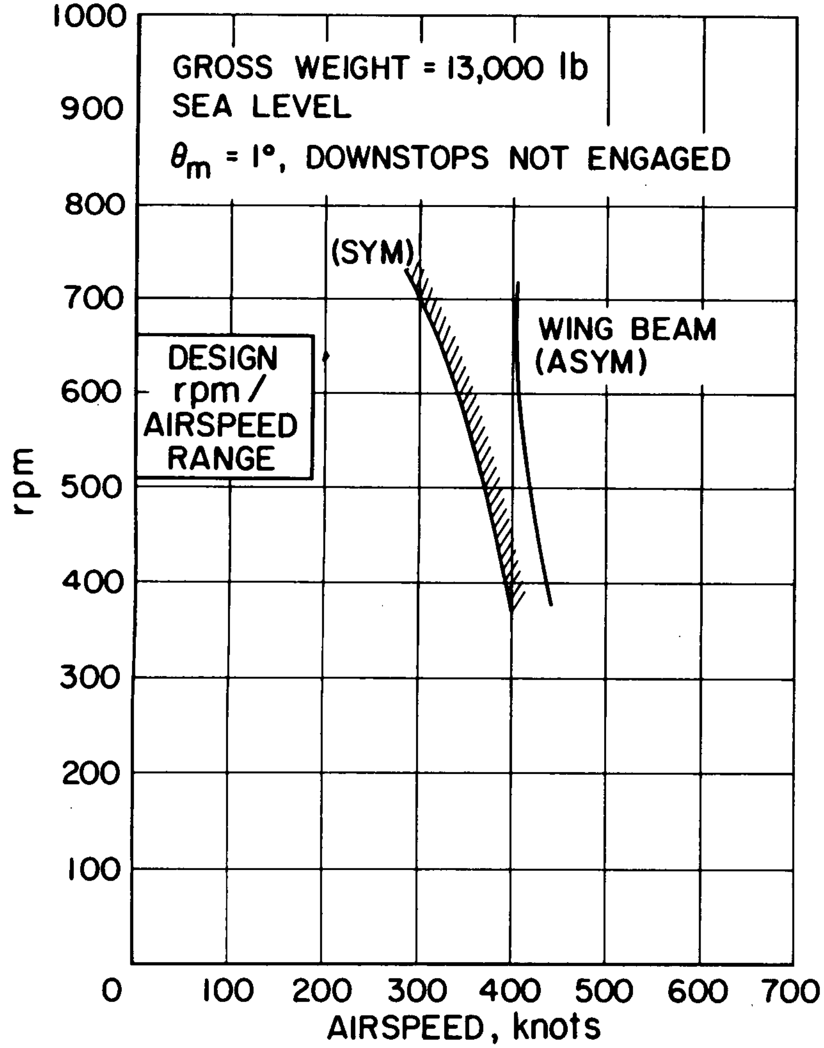


Figure 7.2.1.- Conversion mode stability boundary versus rotor rpm at 1° mast angle.

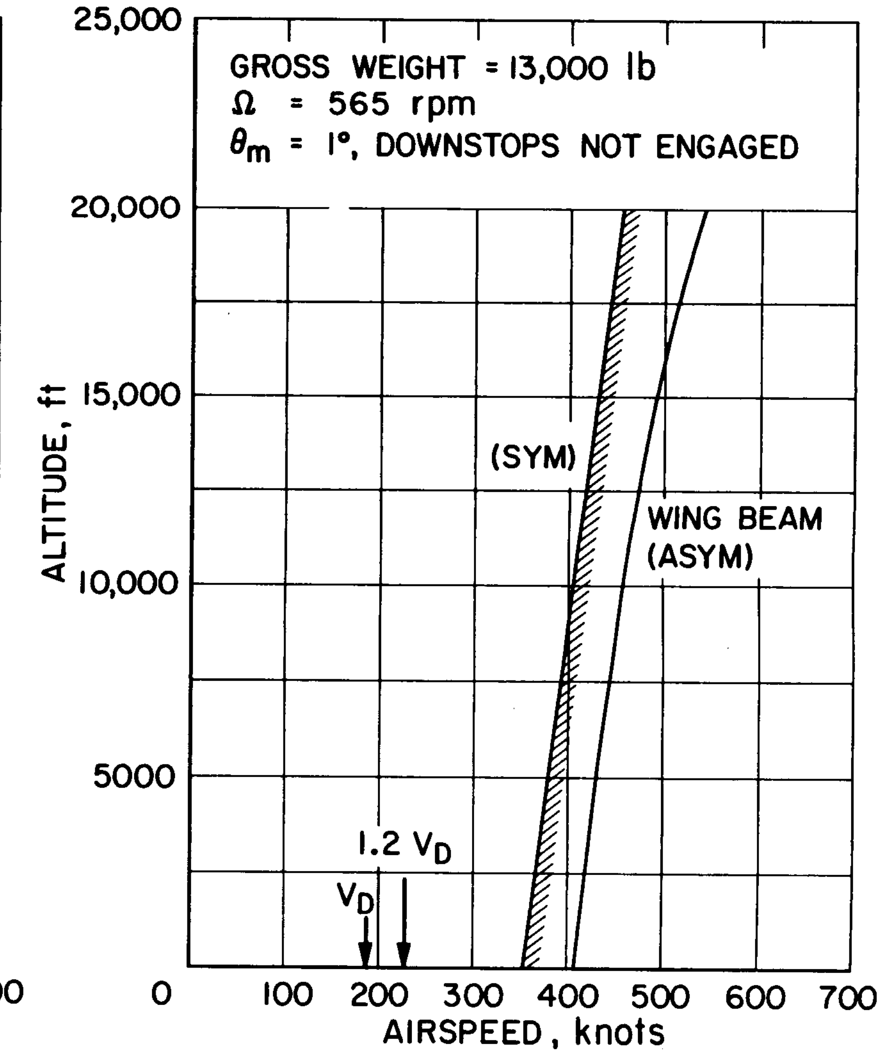


Figure 7.2.2.- Conversion mode stability boundary versus altitude at 1° mast angle.

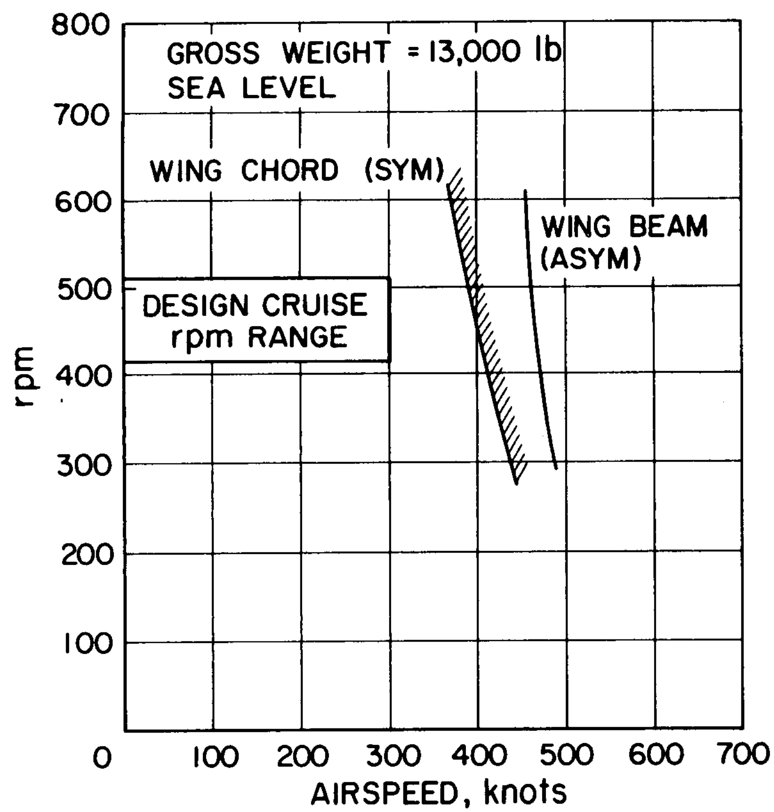


Figure 7.2.3.- Wing/pylon/rotor stability boundary versus rpm, airplane mode.

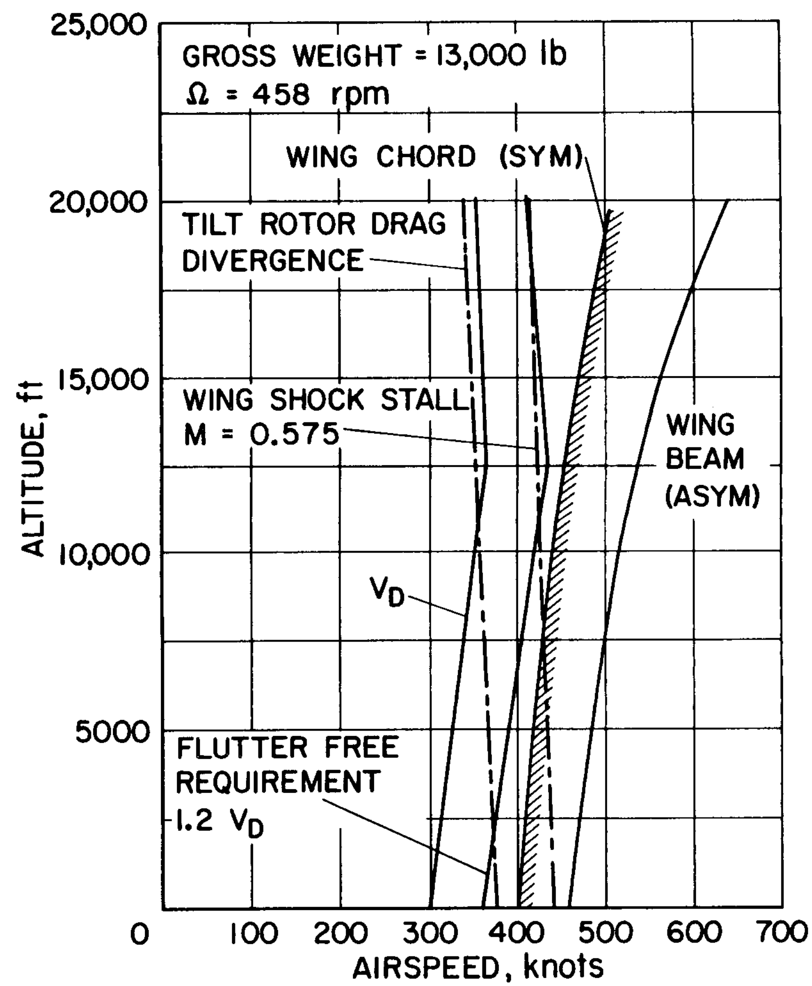


Figure 7.2.4.- Wing/pylon/rotor stability boundary versus altitude, airplane mode.

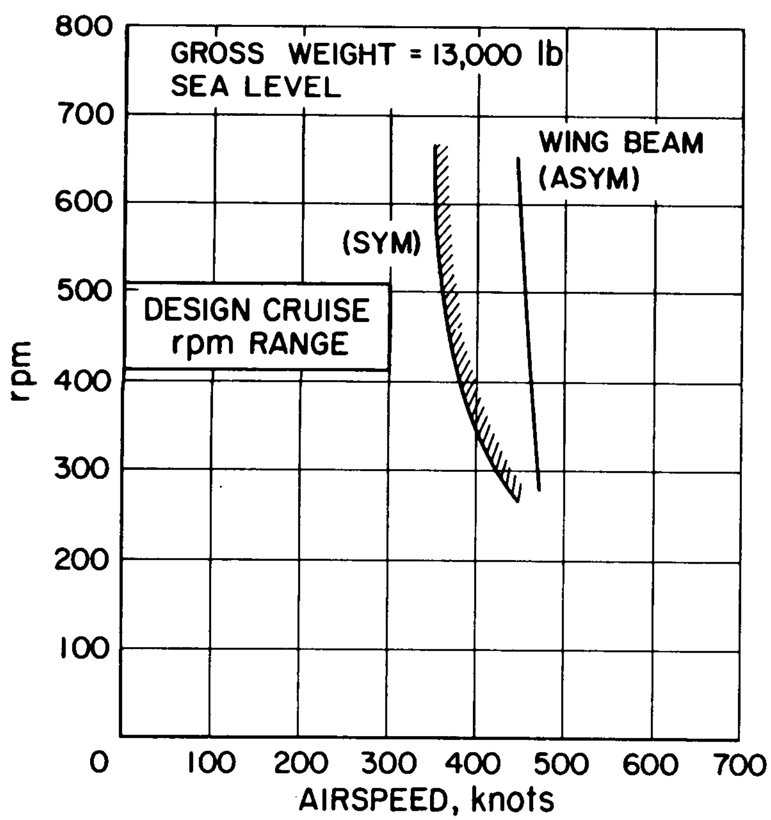


Figure 7.2.5.- Effect of loss of 20 percent of wing and pylon stiffness on rpm stability boundary, airplane mode.

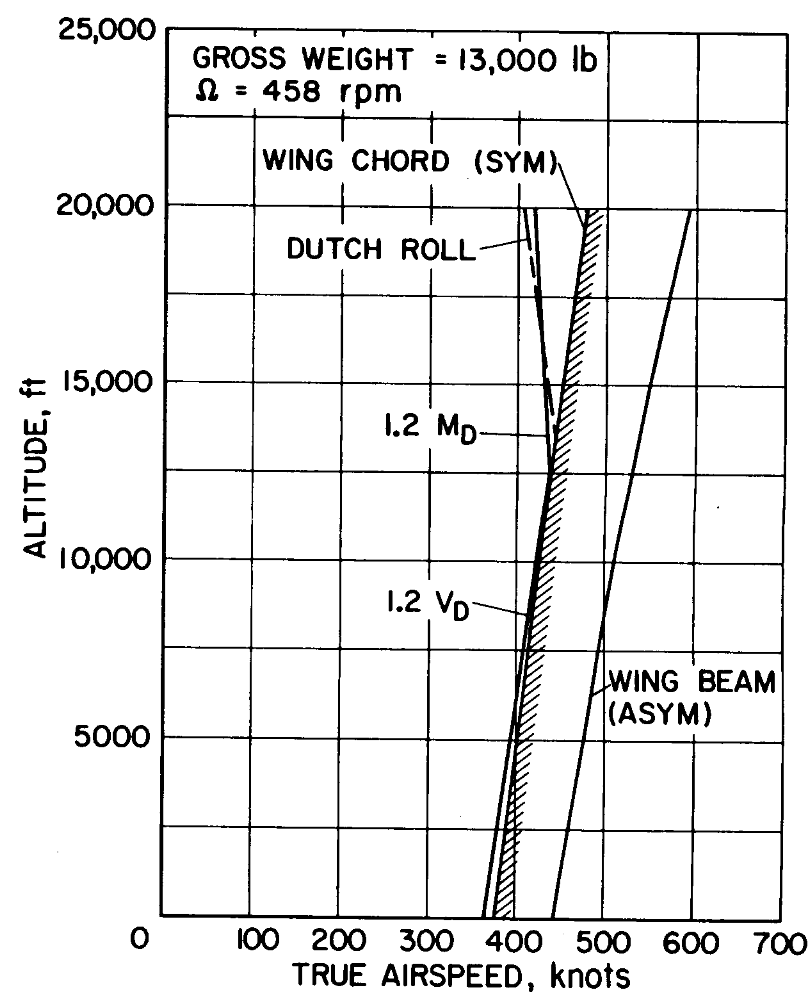


Figure 7.2.6.- Stability boundary versus altitude with 20 percent loss of wing and pylon stiffness, airplane mode.

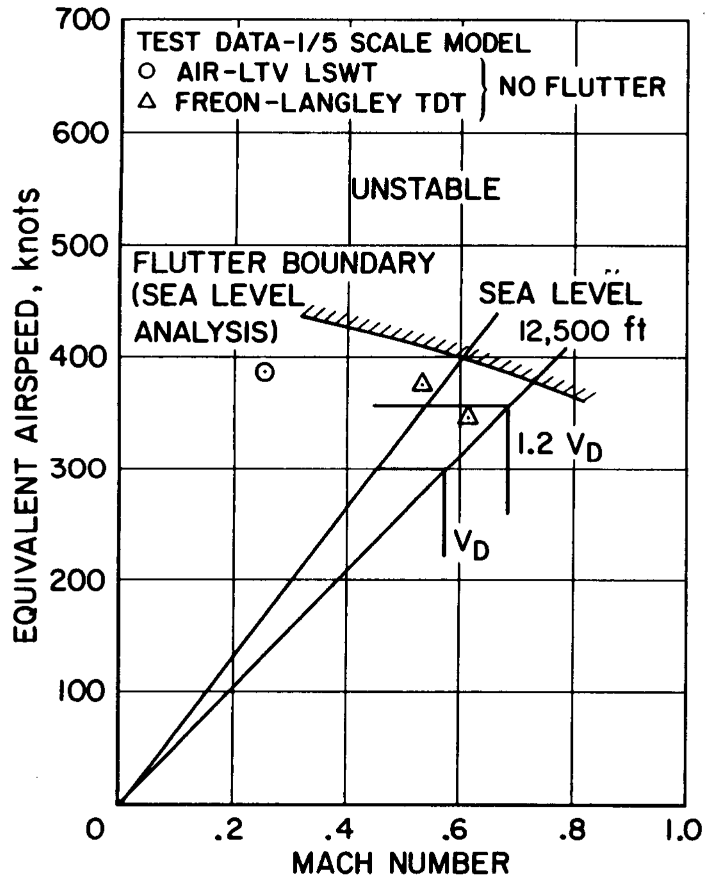


Figure 7.2.7.- Empennage flutter boundary.

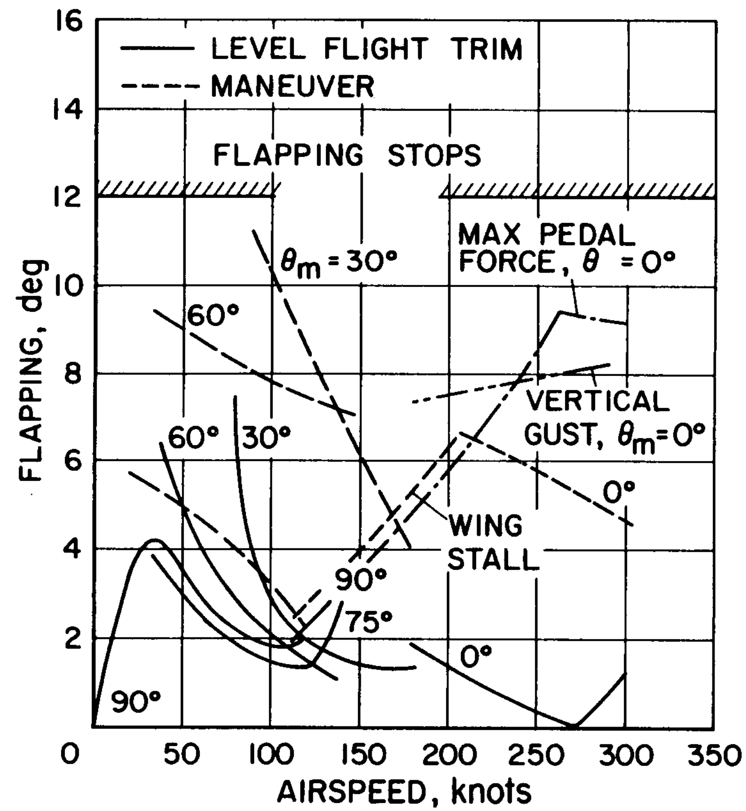


Figure 7.3.1.- Rotor flapping spectrum.

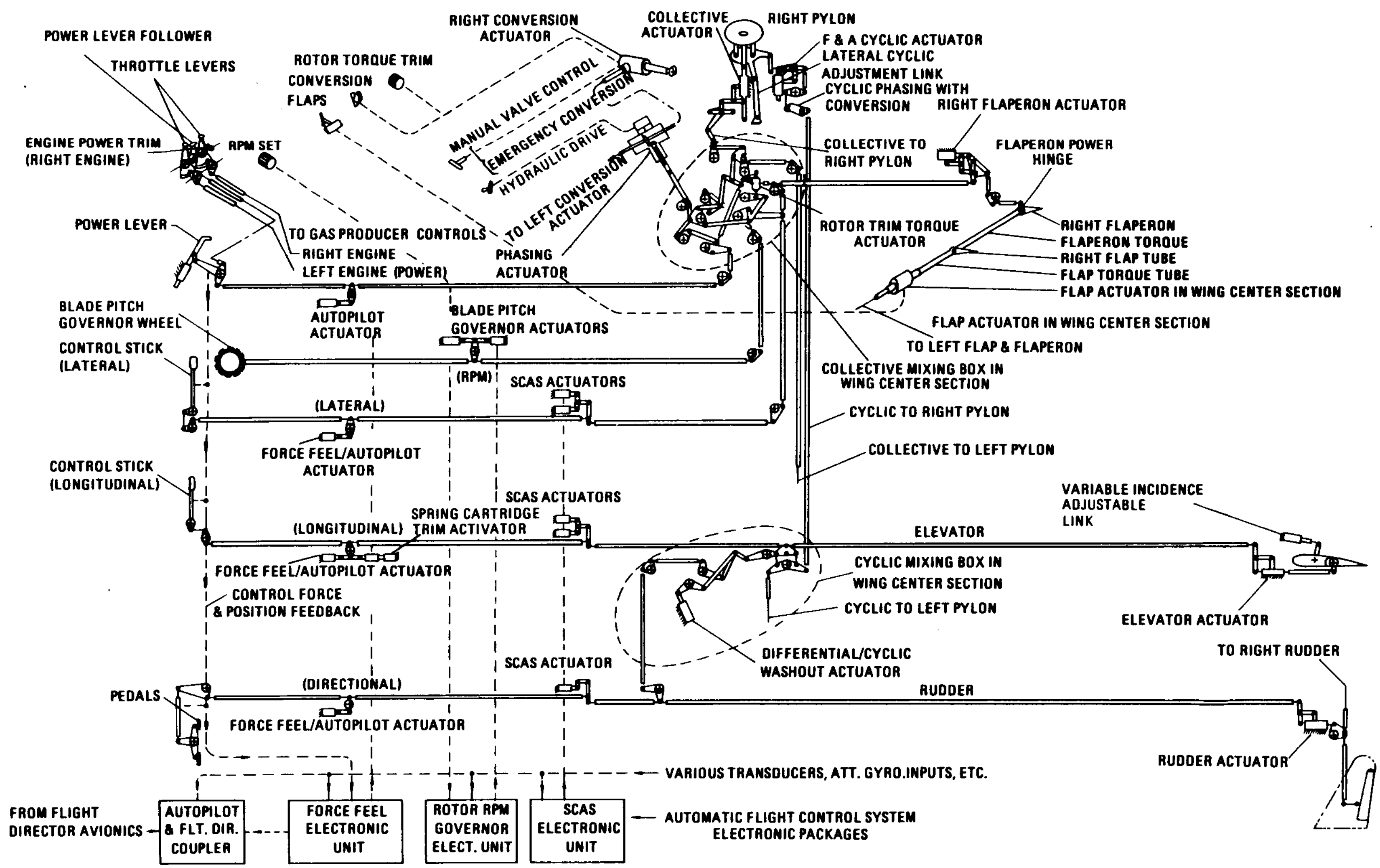


Figure 8.1.1.- Flight control mechanical schematic.

actuators with hydraulic motors and electrically-powered servo valves. These actuators are interconnected by a cross-shaft (fig. 8.1.4.1). Should one actuator fail to function due to hydraulic or electrical failure, that screw jack is driven by the actuator on the opposite nacelle. In the event of a complete electrical power failure, a mechanical backup system, operated by pulling the emergency T-handles located in the cockpit, positions the hydraulic valves to cause the actuators to move the pylons to the helicopter position.

8.1.5 *Power management* - The power management cockpit controls consist of a pair of throttles on the center console and a power lever for each pilot. The collective stick-type power levers are located to the left of each pilot and have the same sense of motion as a conventional helicopter collective stick.

Following engine start and checkout, each throttle lever is hooked to the power lever. Then, in the helicopter mode, power lever motion simultaneously changes the power setting of the rotors. In the airplane mode, however, the power lever only controls power setting of engines as the collective pitch input is phased out as a function of nacelle tilt angle. In addition, power management is simplified by the automatic inputs of the rotor collective pitch governor which adjusts to maintain the rotor rpm selected by the pilot. The rotor governor system is described in section 8.2.3.

8.2 Automatic Flight Control System (AFCS)

The Tilt Rotor Research Aircraft AFCS includes a stability control augmentation system (SCAS), an electrohydraulic force-feel system (FFS), an autopilot with flight director coupler and a rotor rpm governor. Provisions are also included in the control system for a gust and load alleviation system (GLAS).

8.2.1 *Stability control augmentation system* - The SCAS provides attitude retention in all flight modes. It uses three, three-axes-rate gyro packages to sense pitch, roll, and yaw rates, and two attitude gyros to sense pitch and roll attitudes. It also uses two magnetic heading gyros for heading information and three airspeed transducers to provide airspeed retention information and to change gains as a function of speed. Inputs from the conversion angle transducers modify SCAS gains for different conversion angles. This system uses feed forward and feedback loops to tailor the response characteristics and improve static stability.

8.2.2 *Force feel system* - The FFS provides the control forces for the pilot's control in pitch, roll and yaw controls. This system was chosen because of its research flexibility. Unlike a passive mechanical system, this system can produce forces as a function of any given variable. The control force signals are generated electronically and may be rescheduled and modified. The present system configuration provides:

a) gradient shaping with airspeed (lb/in.)

b) damping shaped with airspeed (lb/in./s)

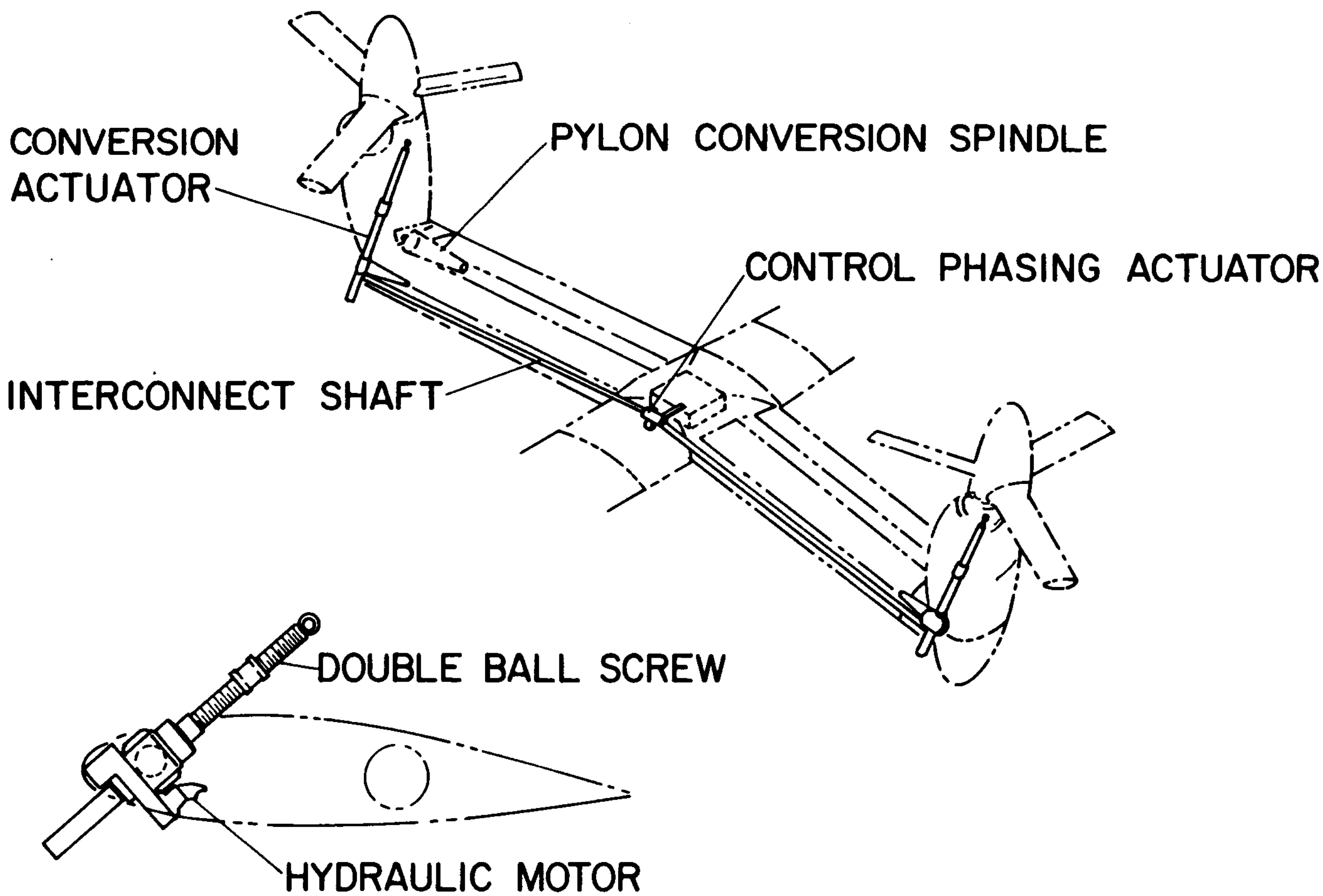


Figure 8.1.4.1.- Conversion system.

c) trim rates as a function of airspeed

d) control system force isolation (SCAS reaction, inertia, friction, etc.)

8.2.3 *Rpm governor* - The rotor rpm governor system is used in all modes to simplify power management. It is a closed loop system that maintains a pilot-selected rpm by controlling collective blade pitch. In the helicopter mode, the collective pitch inputs from the rpm governor are superimposed on the collective pitch inputs from the power lever and the differential collective pitch inputs from the lateral stick. In the airplane mode, the primary collective pitch input comes from the rpm governor as required to maintain pilot-selected rpm. This results from the fact that during transition the collective pitch inputs from the power lever are phased out, and only a small amount of differential collective pitch inputs from lateral stick are retained in the airplane mode to counter adverse yaw. A control mounted on the center console permits the pilot to manually override the rpm governor. If the pilot overrides the system, it automatically disengages.

8.3 Control System Characteristics

Primary control system characteristics are presented in tables 8.3.1 and 8.3.2 and figures 8.3.1 through 8.3.4.

TABLE 8.3.1.- CONTROL TRAVELS

Fore and aft stick	±4.8 in.
Lateral stick	±4.8 in.
Power lever (collective)	10.0 in.
Rudder pedals	±2.5 in.
Pedal adjustment	±2.0 in.
rpm governor wheel	3 turns 8 in. dia.

TABLE 8.3.2.- CONTROL RESPONSE IN HOVER IN 1 SECOND

	Pitch	Roll	Yaw
Control sensitivity (rad/sec^2/in.)			
MIL-H-8501 requirement (VFR)	0.062	0.159	0.165
AGARD 557 criteria	0.06	0.15	0.08
XV-15 SCAS off	0.350	0.471	0.132
SCAS on	0.300	0.489	0.328
XV-3	0.077	0.104	0.048
Control response attitude in 1 second for 1-inch control input (deg/in.$^{-1}$)			
MIL-H-8501 requirement	1.86	1.12[a]	4.55
MIL-F-83300 requirement	3.0	4.0	6.0
AGARD 577 criteria	3.0	3.0	---
XV-15 SCAS off	9.0	6.6	4.0
SCAS on	5.3	11.5	15.0
Control power (rad/sec^2)			
MIL-H-8501 requirement	0.269	0.475	0.475
AGARD 577 criteria	0.40	0.80	0.35
XV-15 SCAS off	1.68	2.27	0.305
XV-3	0.315	0.59	0.042
Damping (1/sec)			
MIL-H-8501 requirement (VFR)	0.452	0.736	1.06
AGARD 577 criteria	0.50	0.50[b]	---
XV-15 SCAS off	0.85	0.77	0.116
SCAS on	1.54	1.82	2.5

gr. wt. = 13,000 lb; aft c.g. @ F.S. 301.2

[a]Attitude in 0.5 sec [b]VTOL with SCAS, damping = 2.0 rad/sec

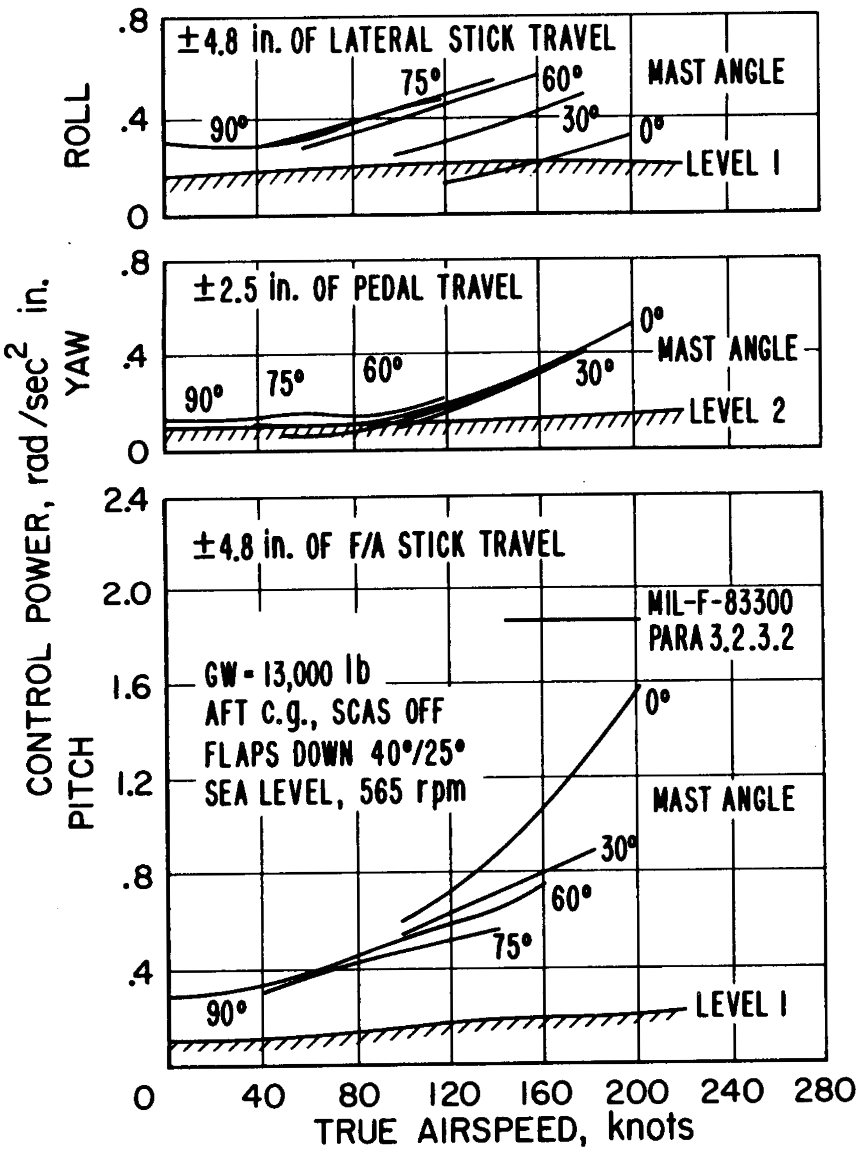


Figure 8.3.1.- Control power variation with airspeed in helicopter and conversion modes.

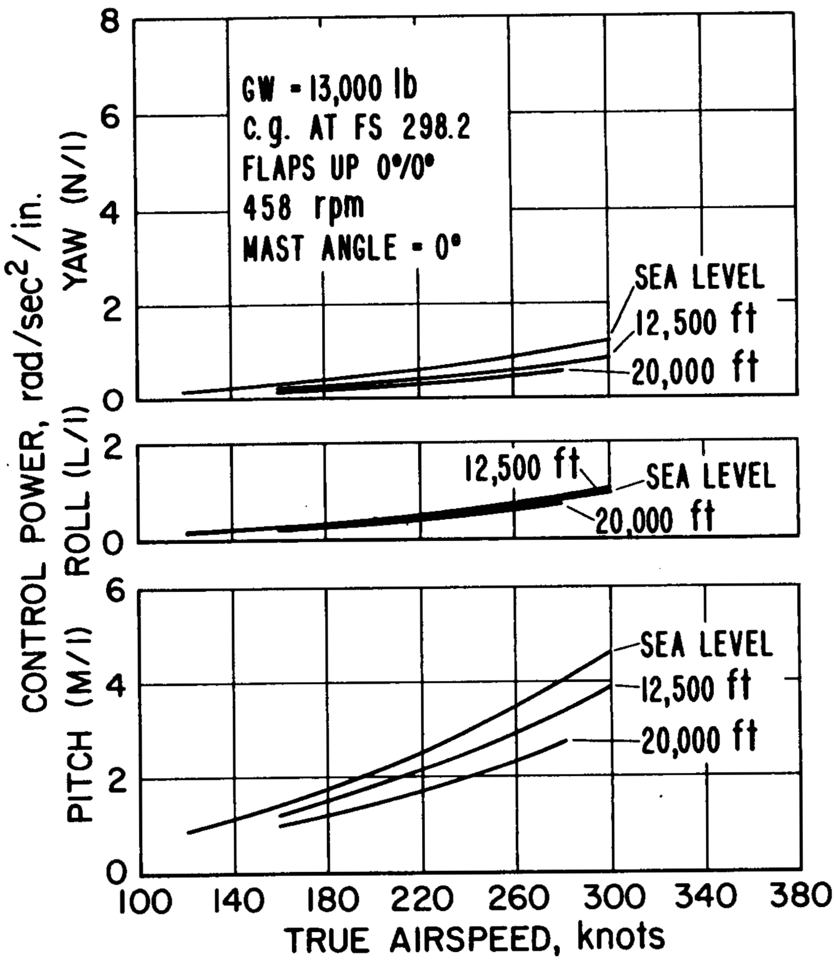


Figure 8.3.2.- Control power in airplane mode.

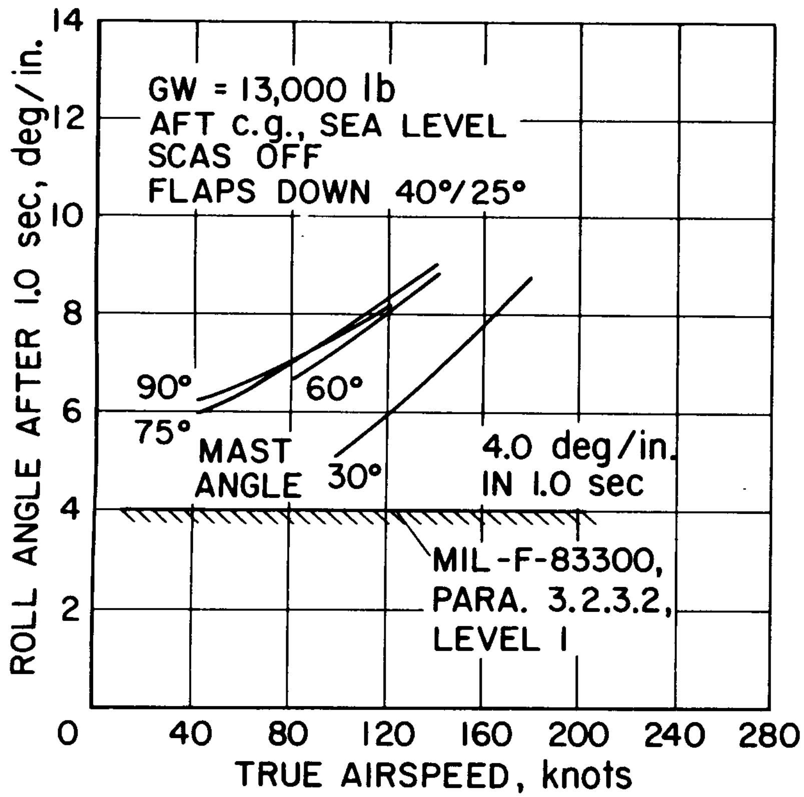


Figure 8.3.3.- Roll performance in helicopter and conversion modes.

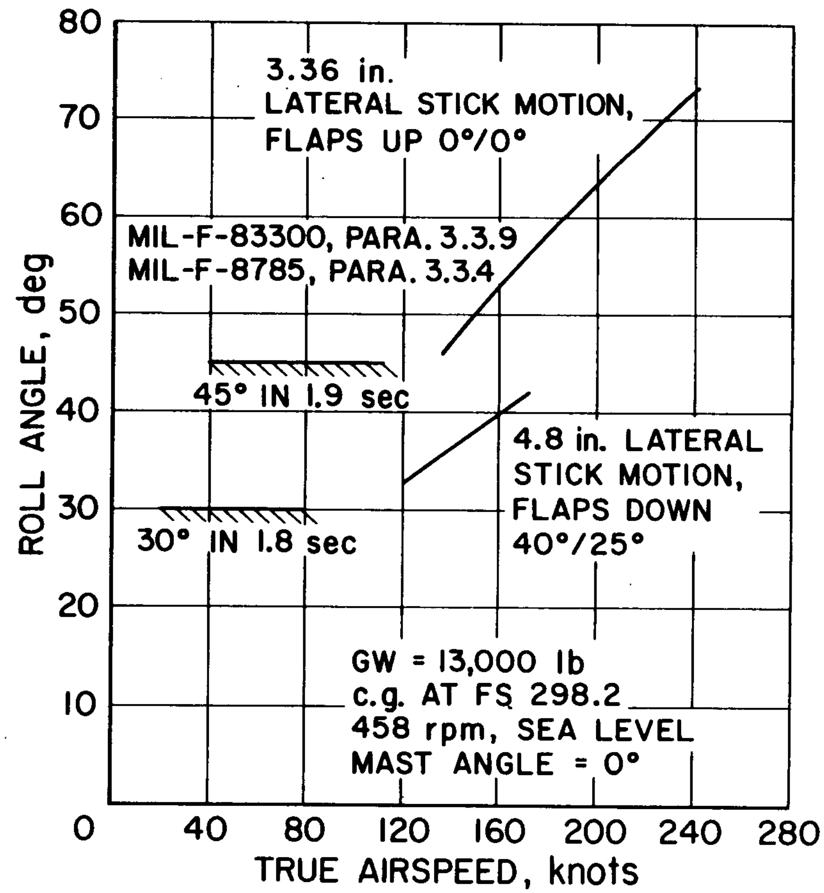


Figure 8.3.4.- Roll performance in airplane mode.

9. RESEARCH INSTRUMENTATION

The research instrumentation system consists of all the instrumentation including transducers, data acquisition system, and tape recorder to measure and record the aircraft systems performance during flight. (Standard flight instruments are defined in section 10.7.1.)

The research instrumentation system, illustrated by the figure 9.1 block diagram, operates as follows. Data from the transducers are forwarded to the remote multiplexer/digitizer unit (RMDU) which provides the signal conditioning for the transducer, adjusts signal gain to programmed value, converts analog data to digital form and encodes the data into a pulse code modulation (PCM) serial bit stream. Transducer excitation (if required) is supplied from a separate low voltage (±3V) power source. Two 64-channel preamplifier filters are available to condition those transducers which require special filtering or amplification. An additional active network panel may be used to condition or process transducer signals whose characteristics do not readily match the interface requirements of the RMDU. A time correlation base for the total system is supplied from a time code generator with a remote time display mounted on the pilot/copilot instrument panel. All data are recorded on a standard airborne magnetic tape recorder. An interface is available for inflight transmission of data from one RMDU via L-band telemetry.

A detailed description of each of the elements in the system block diagram is as follows:

9.1 Transducers

These represent a variety of aircraft sensors to measure aircraft and systems performance control and control surface positions and loads. Table 9.1.1 contains a listing of these sensors and their function.

9.2 Remote Multiplexer/Digitizer Unit (RMDU)

This unit is designed and manufactured by Teledyne Controls. It receives transducer signals in analog or discrete/digital form, conditions/normalizes and multiplexes the input data, converts the analog data to a digital format, and outputs the data in a PCM format. Each unit can accept up to 256 channels of data with a serial output of up to 131,000 words per second. This word rate cannot be utilized by the system due to a tape recorder limitation (single track capacity of 40,000 words per second).

The RMDU is configured for flight by inserting printed circuit cards which interface with the transducers into any of ten card slots. A wide variety of interface cards are available which are compatible with most aircraft transducer signals.

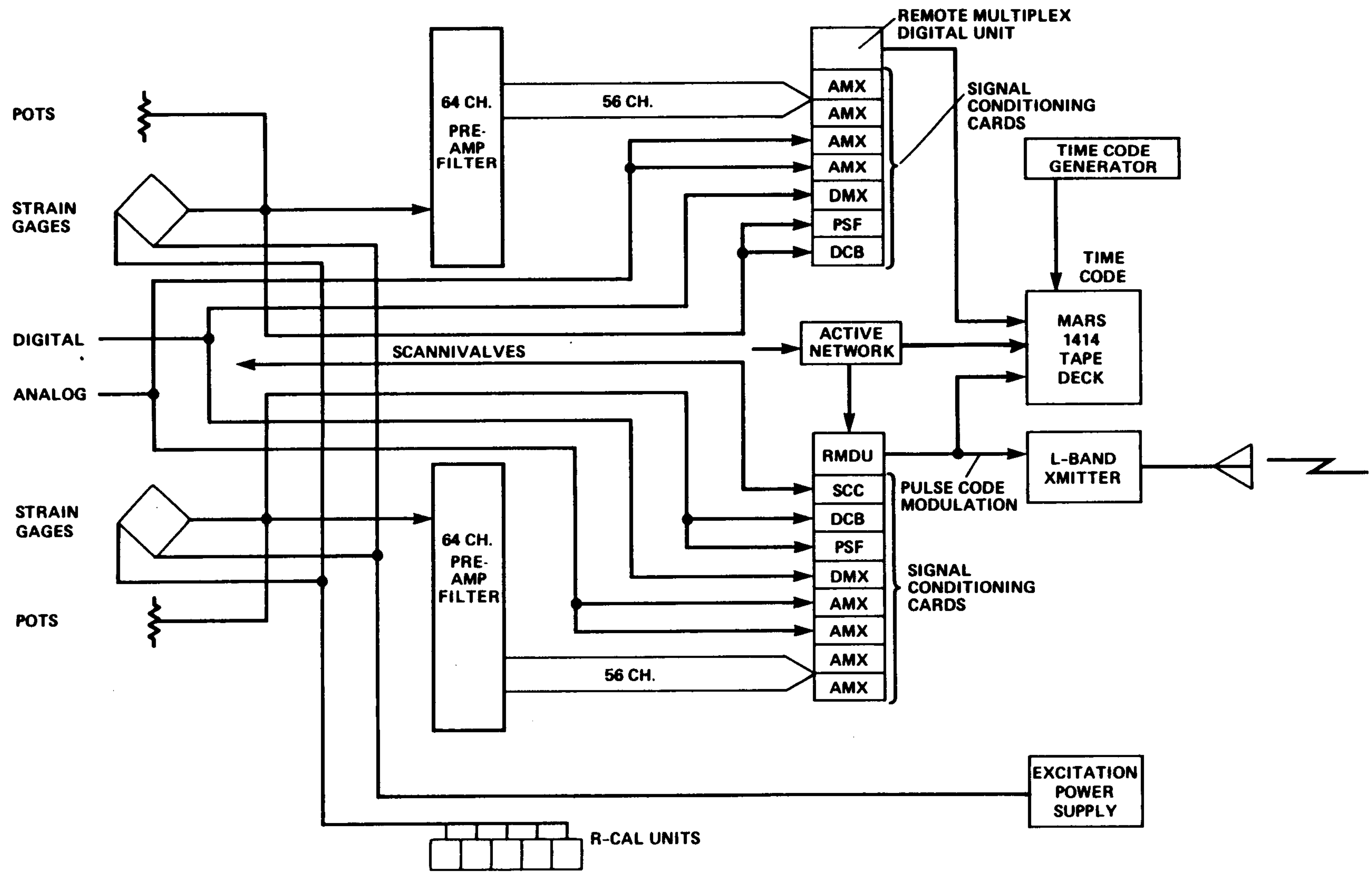


Figure 9.1.- Research instrumentation block diagram.

TABLE 9.1.1.- RESEARCH INSTRUMENTATION TRANSDUCERS

Area	Number of transducers	Description
Propulsion System		
Fuel	2	Total fuel used and fuel flow rate
Inlet	2	Total and static pressure, 17 ports
Inlet	2	Temperature, 4 IC thermocouples
Tailpipe	(to inlet transducer)	Static pressure, 4 ports
Turbine Section	(to inlet transducer)	Temperature, engine T_7 thermocouple harness
Torque	3	Transmission, interconnect
Turbine	2	Gas producer speed (N_I)
Turbine	2	Power turbine speed (N_{II})
Engine vibration	6	Fore and aft, vertical, lateral
Pylon temperatures	(to inlet transducer)	Transmission case and oil, engine oil (15 thermocouples, each engine)
Engine fuel control	2	Fuel control lever position (N_1)
Airframe Loads		
Right wing	2	Upper and lower panel, bending
Right wing	9	Beam and chord bending, and torque
Right horizontal stabilizer	2	Beam bending, upper and lower skin
Right horizontal stabilizer	6	Beam and chord bending, and torque
Right vertical stabilizer	6	Beam and chord bending, and torque
Right pylon conversion spindle	3	Beam and chord bending, and axial load
Right conversion actuator	1	Axial load
Flaps	4	Torque and beam bending
Flaperon	4	Control arm force and beam bending
Elevator	4	Control arm force and beam bending
Rudder	4	Control arm force and beam bending
Pilot Flight Controls, Loads		
Cyclic stick	2	Fore and aft, and lateral
Power lever	1	Force
Rudder pedals	2	Force
Landing Gears, Main, Loads		
Trunnion arm	2	Vertical bending
Oleo strut	4	Fore and aft, and lateral bending
Drag strut	2	Axial force
Nose Gear, Loads		
Trunnion	2	Vertical bending
Oleo	2	Fore and aft, and lateral bending
Drag strut	1	Axial force

TABLE 9.1.1.- RESEARCH INSTRUMENTATION TRANSDUCERS - Continued

Area	Number of transducers	Description
Rotors and Controls, Loads		
Blades	18	Beam and chord bending and torque (R&L)
Hub spindle	4	Beam and chord bending
Masts	6	Bending (2 directions) and torque
Trunnions	2	Fork bending
Pitch link	6	Axial force on all links
Swashplate	2	Drive force
Spinner	4	Support arm beam and chord
Control Boost Actuators, Loads		
Cyclic, F/A	2	Axial force
Cyclic, Lateral	2	Axial force
Collective	2	Axial force
Airframe, Position		
Control surfaces	8	Position measurements
Main landing gear	4	Position measurements
Nose landing gear	3	Position measurements
Horizontal stabilizer	1	Position measurements
Pilot Controls, Position		
Cockpit control	7	Position measurements
SCAS system	3	Position measurements
Rotor Position		
Hub spring motion	4	Position measurements
Gimbal trunnion flapping	2	Position measurements
Blade feathering (single)	2	Position measurements
Collective motion	2	Position measurements
Swashplate motion	4	Position measurements
Conversion motion	2	Position measurements
Rotor azimuth	2	1 per rev
Rotor azimuth	2	512 per rev
Aircraft State Measurements		
Airspeed	1	Pitot port, from nose boom
Altitude	1	Static port, from nose boom
	1	Radar altimeter
Outside air temperature	1	Temperature probe, from nose boom
Relative wind angles	1	Angle of attack, from nose boom
	1	Angle of sideslip, from nose boom
Aircraft attitude	3	Pitch, roll, and yaw
Aircraft angular rates	3	Pitch, roll, and yaw
Vertical acceleration	1	Aircraft c.g.

TABLE 9.1.1.- RESEARCH INSTRUMENTATION TRANSDUCERS - Concluded

Area	Number of transducers	Description
Aircraft System Monitors		
Hydraulic system	3	Pressure
Electrical system	4	dc voltage and current
Fuel system	2	Quantity
Temperature, wing/fuselage	2	Thermocouples A/R to temp. scanner
Oil, engine	2	Pressure
Oil, transmission	2	Pressure
Aircraft Accelerations		
Aircraft center of gravity	3	Fore and aft, lateral and vertical
Pylons	6	Fore and aft, lateral and vertical
Cockpit	4	Lateral and vertical
Test Equipment		
Exciter mechanism	4	Exciter mechanism positions and loads as applicable
Total	222	

Included in the system is a programmable gain amplifier which provides for eight preselected and programmable gains to amplify signal strength to ±5V full scale (gains from ±10MV to 10V full scale).

The RMDU is programmed for frame format, word rate, and gain by a stand-alone-timing module (STM) which is preprogrammed on a ground based PROM programmer. The STM provides PCM outputs of serial NRZ to a telemetering transmitter and serial biphase level to the tape recorder.

9.3 Preamplifier Filter Unit

A preamplifier filter unit is also available for the system. This unit is used for low level signals that require extensive filtering.

The unit provides for 64 channels with a gain from 128 to 1024 and a three-pole active low pass filter.

9.4 Tape Recorder

The tape recorder is an Astro-Science (Bell & Howell) Airborne wide-band FM recorder model MAR S 1414 (LT)-3D. The above unit is a 14-track analog recorder which takes a 14-inch reel of magnetic tape. A PCM bit stream from each RMDU, the time code generator output, and pilot's voice are recorded on separate tracks. The remaining tracks are used to record active network channel outputs if required. The tape recorder is the limiting item in the system due to its bit packing density limitation as related to amount of tape available, speed of recording, and length of record required for flight testing. The above recorder will be operated at 30 inches per second, a record rate of 40,000 words (12-bit words) per second which will allow approximately one hour of full-time data recording. A Miller code will be used to achieve this word rate capacity.

9.5 Time Code Generator

The time code generator is a Datametrics Model SP-375 Airborne Synchronized Generator with integral battery pack which produces an IRIG-B output for recording on the tape recorder and to provide both a local and remotely mounted pilot's display. This unit can be synchronized with radio station WWV (time standard station) and acts as the time base for the research instrumentation system.

9.6 Active Networks

These units are special units which provide contingency options for processing transducer signals which may not be compatible with the RMDU or for special purpose instrumentation requirements. One such example is the thermocouple signal conditioning card.

9.7 Excitation Power Supply

A common power source is provided for excitation of the research instrumentation transducers (other than air data). An Abbott power module supplies ±3V dc for transducer excitation.

10. OPERATIONAL SUBSYSTEMS

10.1 Fuel Systems

Fuel is supplied to each engine by separate fuel systems contained in the wings. Each system has two lightweight crash-resistant fuel cells which are interconnected to form a single tank. The fuel cell bladders, which are continuously supported within the wing by structural honeycomb panels, are constructed of a flexible rupture-resistant material. Total fuel capacity is 1509 pounds of which 19 pounds are unusable. Gravity refueling is accomplished through filler caps in each of the inboard cells.

A submerged boost pump is located at the lowest point of each tank. Each pump is driven by a separate electrical system. Although the two systems normally function independently, gravity and pressure feed interconnect valves and lines provide emergency capabilities, permitting the feeding of both engines from the same tank, or in the case of a single engine failure, one engine from both tanks. A single boost pump failure, sensed by pressure switches, will automatically cause the cross-feed and interconnect valves to be opened, thus providing uninterrupted fuel flow to both engines through common use of the remaining pump. In addition, with a complete loss of electrical power to both boost pumps, adequate fuel flow would be maintained by the engine driven pumps up to an altitude of about 10,000 ft.

A schematic drawing of the fuel system is shown in figure 10.1.1. Fuel passes from the boost pump discharge through a check valve, a shutoff valve, a conversion swivel fitting, a filter, the engine fuel control, and a fuel/oil heat exchanger before entering the fuel flow meter just prior to the engine. A fuel-vapor vent system with a three-dimension siphon break line designed to minimize fire hazard is also provided. A tee fitting is provided to connect an external fuel supply for operation in the Ames 40- by 80-Foot Wind Tunnel.

From the overhead console in the cockpit, the crew can control the fuel shut-off valves, the boost pumps, the cross-feed valve, and the gravity-flow interconnect valve. The caution/warning panel includes indication of right or left fuel-boost pump failure and an indication when remaining fuel quantity falls below 20 percent of total usable capacity.

Manually operated drain valves are positioned on the lower wing surface at each tank sump location and are accessible to ground personnel.

10.2 Hydraulic System

The XV-15 aircraft has three independent transmission-driven 3000-psi hydraulic systems. The pump for each system is geared to the rotor side of the transmission clutch so that full hydraulic power can be provided with both engines shut down, as long as the rotors are turning within the normal speed range. The flight controls are run on "Power Control System No. 2" (PC2) for the single actuators, and on both PC1 and PC2 for the dual- or tandem-actuated components, as shown in table 10.2.1. The third hydraulic system (PC3) normally powers the landing gear actuators, a nacelle heat exchanger blower and aerodynamic excitation actuators. Upon the loss of PC1 or PC2, automatic shut-off valves isolate the excitation and landing gear actuators from the PC3 system, which is then held in reserve as a backup for some critical PC2 items. If the PC2 system pressure drops below 1000 psi, automatic shuttle valves will provide PC3 fluid to the PC2 side of the flight-critical collective pitch and flaperon tandem actuators.

A pneumatic backup system is used to deploy the landing gear after PC3 becomes dedicated to the critical flight controls. This subsystem's pressurized nitrogen reservoir is connected to a shuttle valve on the inlet ports of each landing actuator. Deployment of the landing gear in the pneumatic

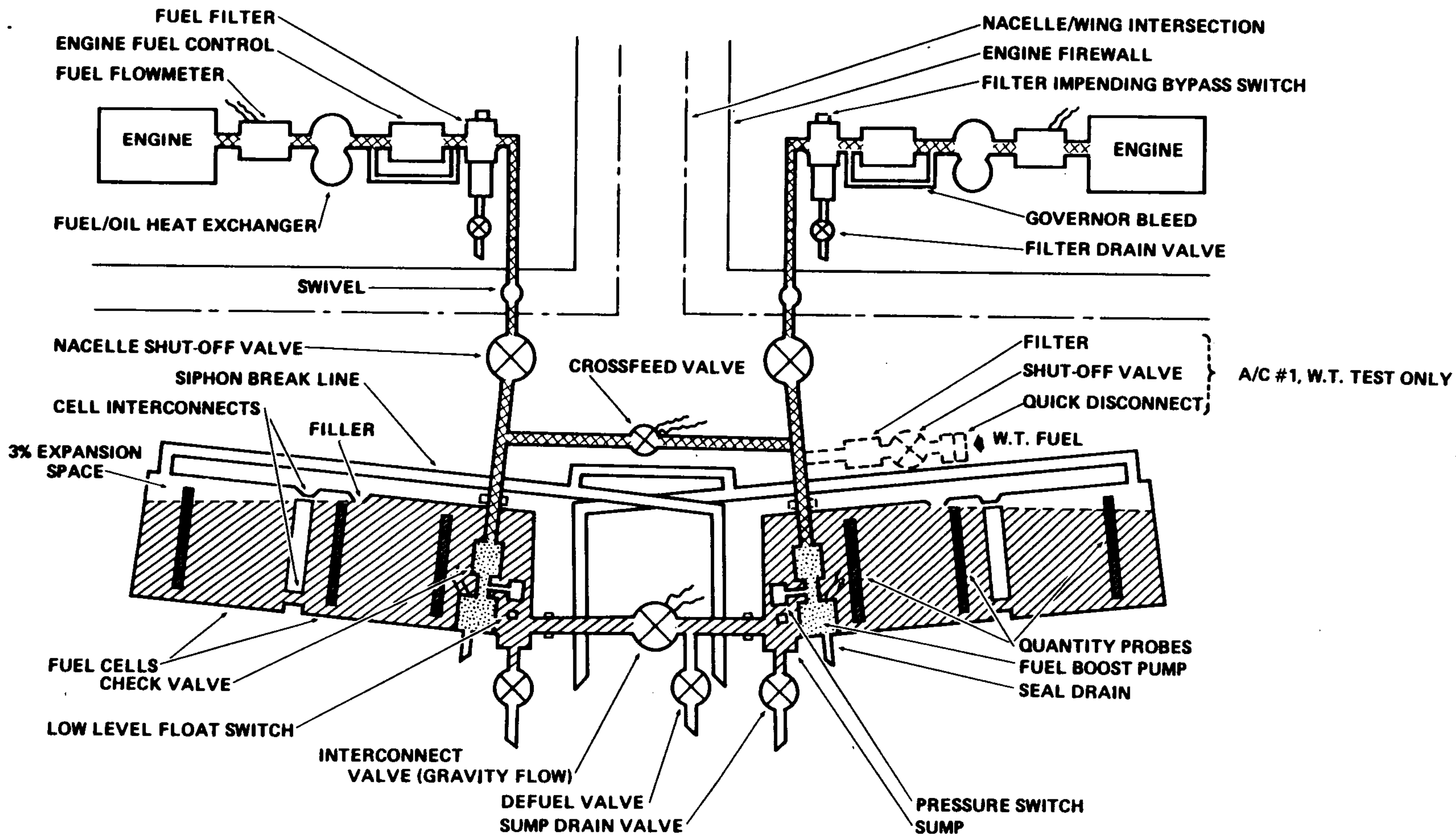


Figure 10.1.1.- Fuel system schematic.

TABLE 10.2.1.- HYDRAULIC POWER DISTRIBUTION

Component	Type	Subsystem: PC 1	PC 2	PC 3
		Swivel location		
		outboard (pylon)	inboard (pylon)	inboard (spindle)
		Pump location		
		L/H transmission	R/H transmission	L/H transmission
Fore/aft cyclic	T	X	X	
Elevator	T	X	X	
Flaperon	T+	X	X	X_1
Rudder	S		X	
Lateral cyclic	S		X	
Excitation	S			X_2
Conversion	D	X	X	
Collective	T+	X	X	X_1
RPM governor	T	X	X	
SCAS - Pitch	D	X	X	
Roll	D	X	X	
Yaw	S		X	
Force Feel -				
Pitch	S		X	
Roll	S		X	
Yaw	S		X	
Pitch Trim	S		X	
Emergency conversion	Motor			X
Heat exchanger blower	Motor		X	X
Landing gear	S			$X_{2,3}$

Remarks:

Type

- T Tandem actuator
- T+ Tandem actuator with shuttle valve to permit PC_3 backup for PC_2
- D Dual actuator
- S Single actuator

Subscripts:

1. Employed upon loss of PC_2
2. Isolated from PC_3 upon loss of PC_2 or PC_1
3. Pneumatic backup employed after loss of PC_1, PC_2, or PC_3.

backup mode is initiated by the crew through a mechanical linkage between the crew station and the pneumatic shut-off/actuation valve.

A schematic of the XV-15 hydraulic system is presented in figure 10.2.1. Each of the three PC subsystems is a closed center type with an airless pressurized reservoir, filters, a variable displacement pressure compensated pump, a heat exchanger, actuators and hydraulic motors, valving, conversion swivel joints, and interconnect tubing and fittings. Ground test and fill provisions are included for each system with the quick-disconnect fittings located in the left main landing gear pod. Onboard filters at these quick-disconnect points prevent contamination from external fluid sources. A tee fitting is provided in the wing for connecting PC2 to a hydraulic power source in the Ames 40- by 80-Foot Wind Tunnel.

The hydraulic system is designed to use the standard MIL-H-5606 aircraft hydraulic fluid. Pump capacity at the design helicopter rotor speed is about 7.6 gallons per minute for PC1 and PC3, and about 13.0 gallons per minute for PC2.

Temperature and pressure monitoring gauges for the flight control hydraulic systems (PC1 and PC2) are included on the instrument panel. In addition, the caution/warning panel indicates failure any of the three hydraulic systems with amber lights. Low pressure, high temperature, or low quantity will cause the warning to be illuminated for the affected system. Preflight checks of the hydraulic systems are controlled with overhead console switches which deenergize any selected system in order to functionally test the remaining lines. The PC3 emergency conversion switch and the conversion motor declutch and assymetric conversion bypass switches are also located on the overhead panel.

The hydraulic braking system on the dual main landing gear wheels is completely independent of the PC1, PC2, and PC3 hydraulic systems.

10.3 Electrical System

Figure 10.3.1 is a block schematic of the XV-15's electrical system. The system consists of dual dc and ac electrical subsystems with sufficient capacity to accommodate peak load requirements with one engine out. Extensive use of airframe grounding is made for circuits that are not dependent on extremely low noise levels. Overload protection is provided for all electrical circuits.

10.3.1 *Dc System* - Primary dc power is generated by two 300-ampere starter-generators, one located on each engine. Current from these generators is distributed to the two 28V dc busses. These busses are normally independently supplied from their respective generators and are electrically and mechanically isolated from each other.

In the event of a dc generator failure, or a failure of the bus feeder wires, automatic voltage sensing circuitry will immediately remove the affected generator from its bus and connect the two dc busses in parallel. This

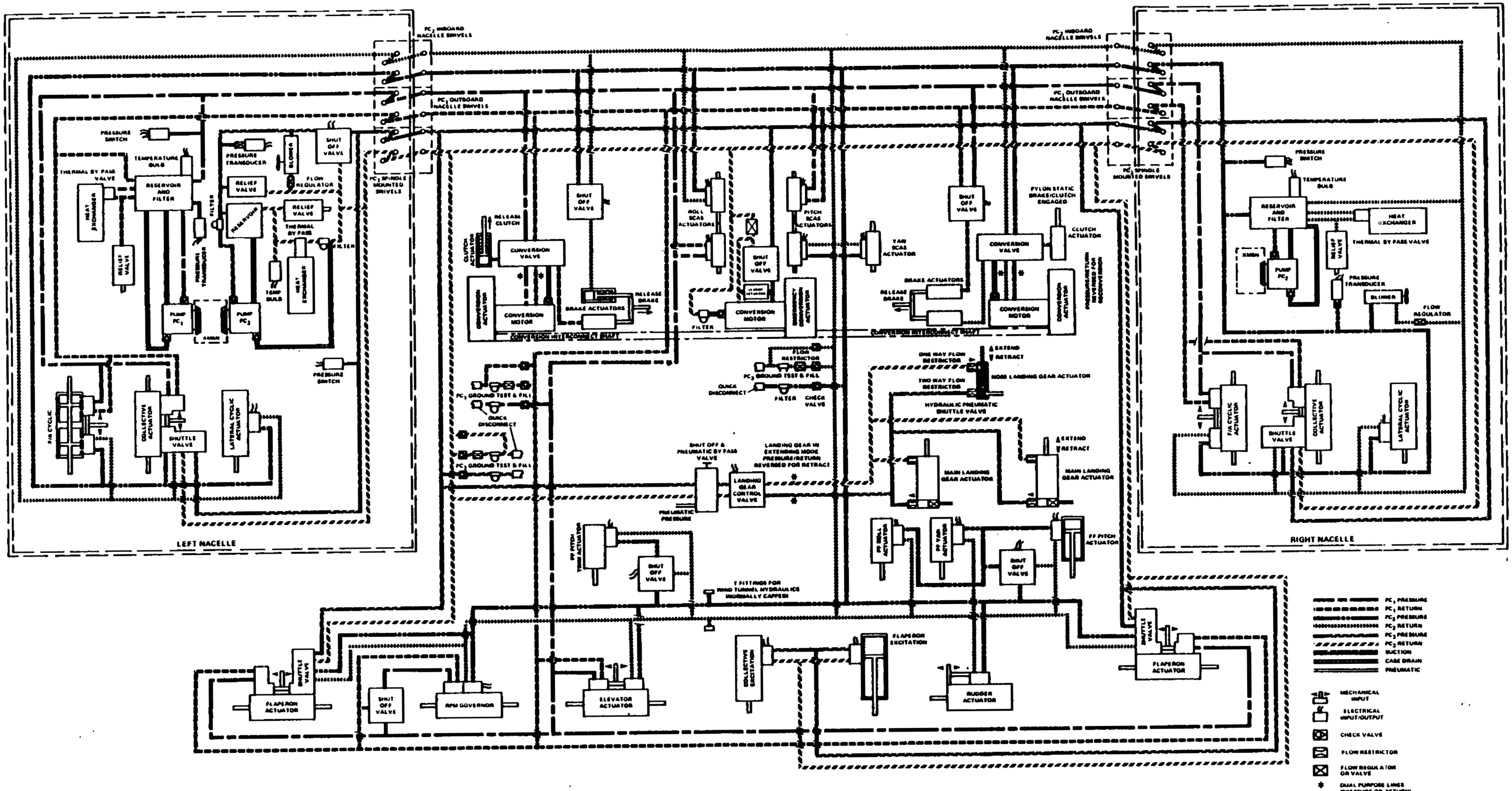
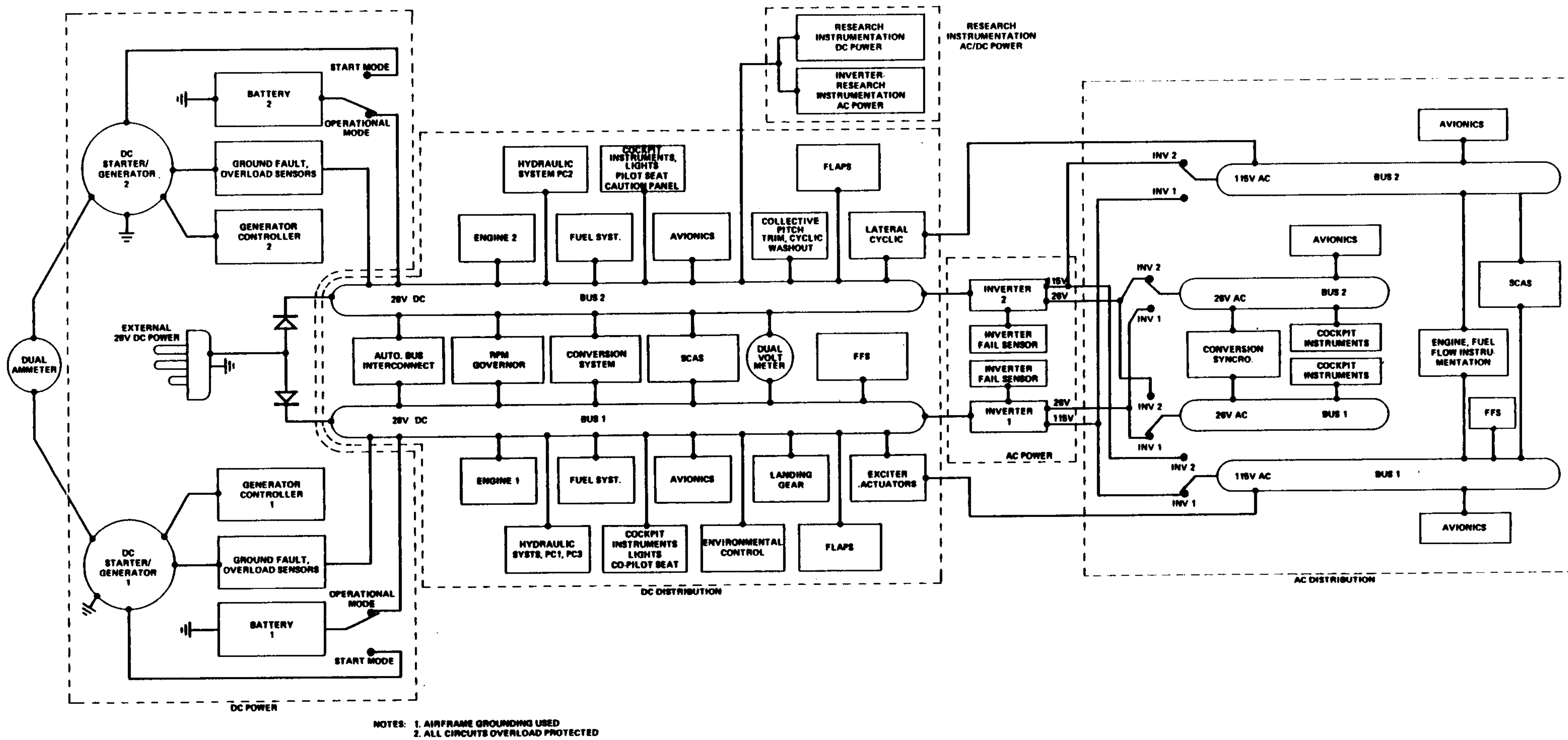


Figure 10.2.1.- Hydraulic system schematic



ORIGINAL PAGE IS
OF POOR QUALITY

Figure 10.3.1.- Electrical system block diagram.

PRECEDING PAGE BLANK NOT FILMED

FOLDOUT FRAME 1

FOLDOUT FRAME 2

provides immediate restoration of power to the affected bus. Thereafter, the remaining dc generator will continue to supply dc power to both dc busses. Each generator can provide the full XV-15 electrical power requirement. A current sensor between the two busses will limit current flow to protect the unfailed system in the event of a fault on one bus.

A 13-ampere-hour battery is connected to each dc bus during normal operation, so that bus voltage should not drop below 24 volts during bus switching operations. The batteries provide a self-contained engine-start capability.

The automatic bus interconnection circuitry is provided with manual override switches on the crew station overhead consoles so that the pilot may manually connect or disconnect the dc busses at any time. Manual switching of the dc generators may be done from the overhead console as well.

Emergency dc power will be supplied to the individual busses by the two batteries in the unlikely event of failure of both dc generators. On-off switches for the two batteries are also located on the electrical switch panel.

10.3.2 *Ac System* - Ac power on the aircraft is provided by two 600 VA solid-state inverters. One inverter is powered by dc bus 1, the other inverter is powered by dc bus 2.

These inverters supply both 26V and 115V ac current at 400 Hz, with a distribution bus connected to each output. The busses are normally independently supplied from their respective inverters, and are electrically and mechanically isolated from each other.

In the event of an inverter failure, voltage sensing circuitry will cause the master caution lights to illuminate, and a light in the caution panel will advise the pilot of an inverter failure. The pilot may then switch off the affected ac bus system and select ac power from the remaining inverter. There is a switch associated with each ac bus system which allows the busses to be powered from either of the two inverters, or, when placed in the "OFF" position, will remove all ac power from the busses.

With the present electrical load, one inverter can supply all of the ac power requirements of the aircraft.

10.3.3 *Research Instrumentation Power* - 28 volts dc and single phase, 400 Hz, 115 volts ac electrical power are supplied for the research instrumentation. A separate solid state inverter on the number two main dc bus supplies 750 VA, 115 volts. Protection and isolation is provided so that the aircraft power system is not affected by failures in the research instrumentation, which is in turn protected from aircraft induced transients.

10.4 Environmental Control System

The environmental control system provides crew station heating, ventilating, air conditioning, window defogging and crew breathing oxygen.

10.4.1 *Temperature Control and Ventilation* - The heart of the temperature control and ventilation system, schematically illustrated in figure 10.4.1.1, is the AiResearch air-cycle environmental control unit (ECU) which uses right-engine-bleed air as a power and hot air source for the heating or cooling operational modes. An ambient air-inlet diverter valve enables the introduction of unconditioned air for fresh air ventilation of the crew station. An electrically powered inlet fan provides the required air flow at all flight conditions. Insulated polycarbonate ducts are used for the delivery of the conditioned air to the crew station. Distribution is controlled by the crew through the use of adjustable outlets at the left- and right-hand corners of the instrument panel and by side-window defog/torso air control valves. Windshield defog outlets provide continuous air flow whenever the ECU or ambient air inlet fan are operating. A return air duct from the crew station to the ECU reduces the temperature rise that would be encountered if the ECU air inlet accepted uncooled air from the aft cabin. About 50 percent of the air processed by the ECU is cockpit return air. On hot days, a curtain at the cockpit bulkhead may be closed to reduce the loss of cooled air to the unoccupied aft cabin.

Noise control is achieved by the incorporation of a noise suppressor unit in the conditioned air distribution system (to attenuate turbine noise) and by acoustic treatment of the return duct (to reduce compressor noise).

In addition to the air vents, the crew can regulate the cabin environment from controls on the overhead console. On-off switches for the ECU and the ambient air fan are provided. A temperature control is also mounted on the ECU panel.

10.4.2 *Oxygen* - Each of the two separate oxygen systems illustrated in figure 10.4.2.1 consists of an 1800 psi oxygen cylinder, mounted forward of the cockpit, which supplies one crew member oxygen through a diluter-demand-type oxygen regulator. The pilot and copilot oxygen control panels on the side consoles provide a flow indicator, a cylinder pressure gauge, an on-off switch, test capability, and a bypass switch to allow 100 percent oxygen to be delivered to the face mask, if required. An emergency switch maintains a positive pressure at the mask to prevent contaminated cabin air from entering. The flexible line from the face mask attaches to a coupling on the lower surface of the regulator unit. The cylinders contain adequate oxygen for a one-hour flight at 100 percent oxygen with a 20 percent reserve.

Access doors are provided in the nose of the aircraft to enable preflight inspection of the pressure gauges mounted on the cylinders and to allow ground refill and servicing. A single point filler valve is used to fill both cylinders simultaneously.

10.5 Communication/Navigation and Flight Director Systems

10.5.1 *Avionics* - The XV-15 avionics system design and integration employs concepts formulated through experience gained with the avionics systems installed for the U. S. Army and other services. The avionics system utilizes off-the-shelf components with little or no modification.

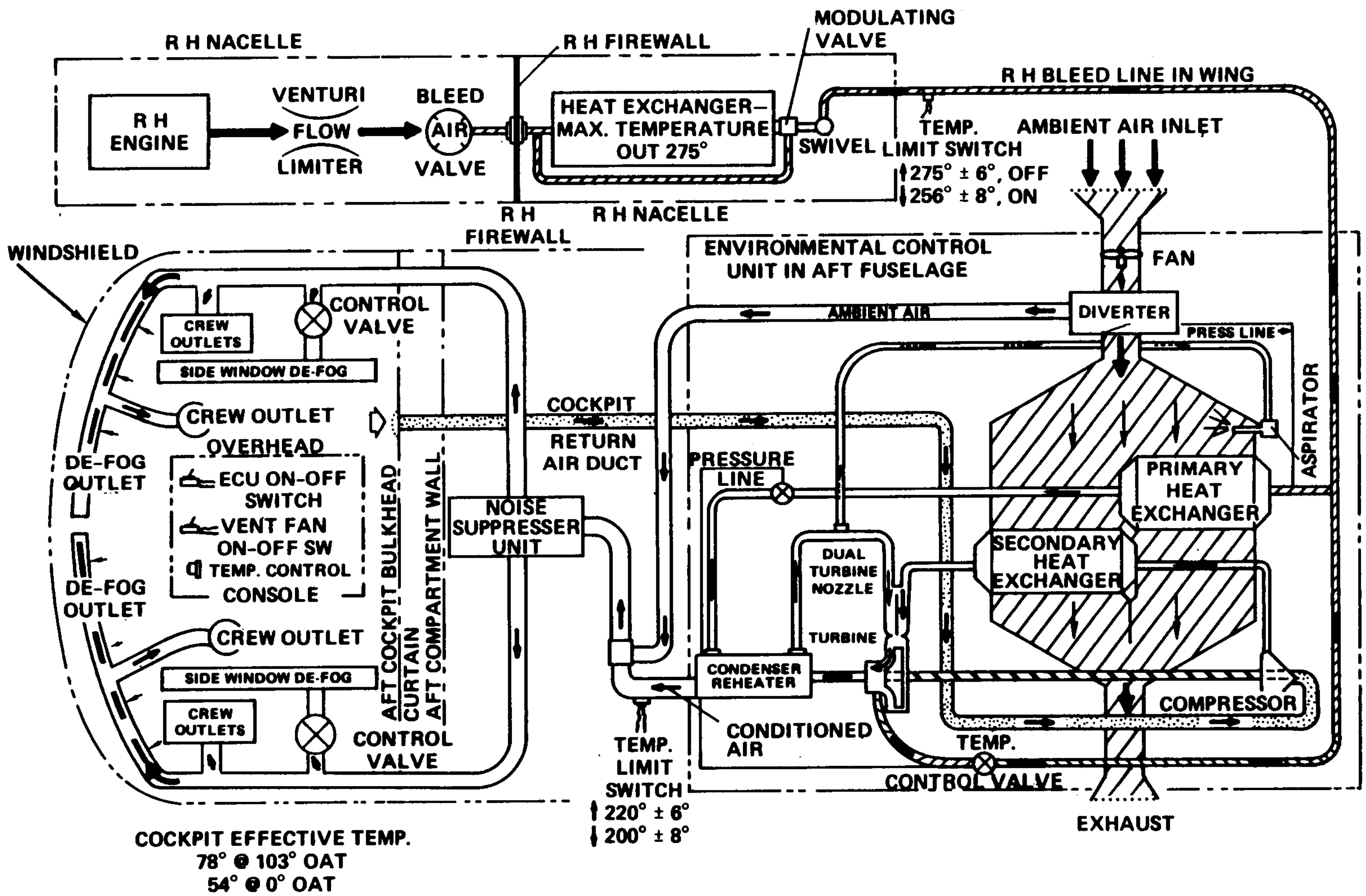


Figure 10.4.1.1.- Environmental control system.

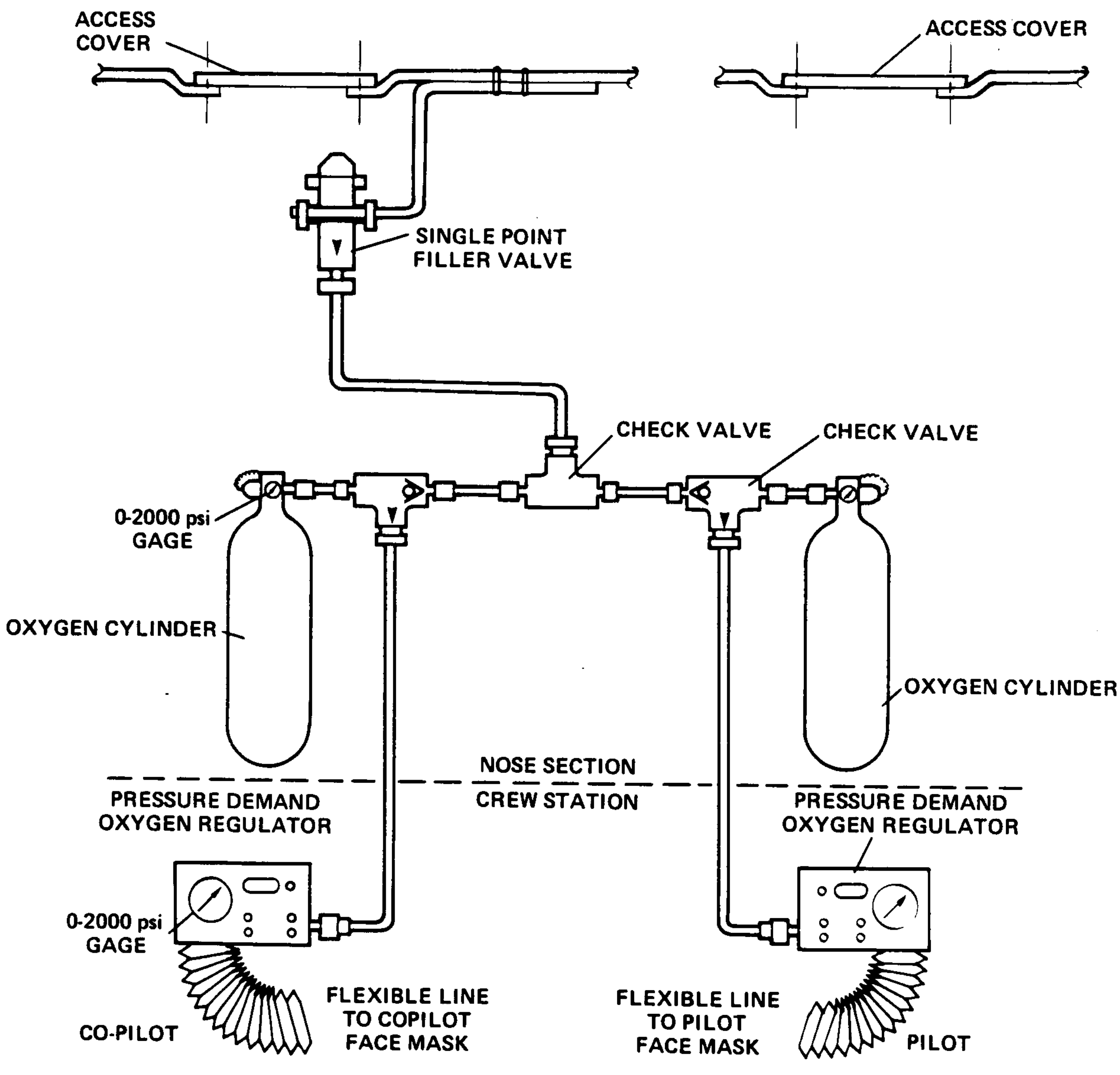


Figure 10.4.2.1.- Oxygen system schematic.

10.5.1.1 *Communication system* - The communication system consists of the following components:

a) One VHF (King KTR-905), amplitude modulated, lightweight airborne radio which transmits and receives voice communication in the frequency range between 118.00 and 135.975 MHz and 25 kHz channel spacing (720 channels). This radio set consists of a transmitter-receiver, control head, and an antenna. The transmitter and receiver are located in the aft avionics compartment and the control is mounted in the center console, accessible to the pilot and copilot.

b) One UHF radio (AN/ARC-51), amplitude modulated airborne radio with 3500 discrete channels in the frequency range between 225.00 and 399.95 MHz. This radio also provides an independent fixed-channel "guard" receiver tuned to 243.00 MHz, the UHF emergency frequency. The radio set consists of a transmitter-receiver, control head, and an antenna. The transmitter-receiver (KT-742/ARC-51BX) is mounted in the aft avionics compartment. The control head is mounted in the cockpit and is accessible to the pilot and copilot.

c) Two intercom systems (C-6533/ARC) provide a complete radio/navigation control system. This set controls all verbal communications, both internal and external to the aircraft. The pilot's and copilot's controls are located on the side and center consoles. They provide individual transmitter selection, independent receiver audio monitoring, navigation receiver selection, and permit simultaneous transmit-interphone talk. Foot operated press-to-talk switches are installed for convenient operation by the pilot and copilot. A rocker-type press-to-talk switch is installed in each cyclic control grip. Pressing the switch to the right, keys intercommunication, and pressing the switch to the left, keys the radio selected on the rotary-type transmitter selection switch of the C-6533/ARC control. The intercommunication control also provides an external jack for pilot/ground crew hookup during ground operations.

10.5.1.2 *Navigation system* - The navigation system consists of the following components:

a) VOR/ILS Navigation - The King KNR-630 is a lightweight, self-contained, console-mounted receiver system that provides VOR/LOC, and glideslope indications to the horizontal situation indicators and attitude direction indicators. This system also automatically channels the distance measuring equipment (DME). The receiver is tunable from 108.00 to 117.95 MHz in 200 channels at 50 kHz spacing. The glideslope receiver covers the frequency range of 329.3 to 335.0 MHz in 40 channels. The glideslope channel is automatically selected when the localizer frequency is tuned. When coupled with the flight director system, the set presents command steering information to the attitude director indicator. A King KMR-675 marker beacon receiver is incorporated in the VOR/ILS system that activates marker beacon lights on the panel.

b) Distance Measuring Equipment (DME) - The King KDM 705 DME is a distance/ground speed/time-to-station measuring unit operating at altitudes up to 30,000 ft. Channeling of the DME is accomplished automatically by the King KNR 630 VOR/ILS system. The receiver-transmitter is a self-contained unit mounted in the aft avionics compartment with the antenna mounted to the underside of the fuselage. Only the distance measurement capability of the DME is used, and this is displayed on the horizontal situation indicator (HSI).

c) Gyro compass - A Sperry C-14 gyro-magnetic compass system is used to provide navigation information to the copilot's radio magnetic indicator, the pilot's HSI, and the autopilot system. The C-14 compass system consists of a direction gyro (P/N 2587193-1) located on the avionics rack, a magnetic flux compensation and compass indication transmitter (both located in the tail section), and a compass control panel located in the pilot's (right) side console.

d) Radar Altimeter - A Honeywell HG-7602AC01 precision pulse, leading edge tracking radar altimeter is installed. The receiver-transmitter and indicator are mounted as a single unit in the pilot's instrument panel. There are also provisions for recording radar altitude on the research instrumentation system.

This radar altimeter can operate from 0 to 5000 ft and has a track rate capability of ±2000 ft/s. An on-off switch on the face of the instrument has an altitude "decision height" setting. A light on the instrument informs the pilot when the preset low altitude is reached. In addition, a built-in feature of this radar altimeter will activate the landing gear warning light whenever the radar altitude is less than 200 ft with the landing gear retracted.

The LG 81A antennas are flush mounted on the underside of the aft fuselage.

10.5.2 *Flight Director System* - One flight director system (FDS) will be installed and consists of the following:

a) one flight director computer (FDC)
b) one horizontal situation indicator (HSI)
c) one attitude direction indicators (ADI)

The flight director computer (FDC) computes navigation commands for presentation on the attitude direction indicator located in the pilot's panel. It also provides navigation mode and equipment switching in one self-contained package. The FDC uses radio navigation, sensed flight situations, and magnetic headings to command selected modes. The FDC provides properly damped command outputs for the automatic flight control subsystem for three-cue operation, the following navigation modes are available:

a) intercept and flight along a magnetic heading
b) intercept and flight along a VOR radial
c) intercept and approach to an ILS localizer

The horizontal situation indicator (HSI) is a multifunction indicator providing the pilot with a complete navigation information presentation. The HSI presents relative magnetic bearing information from the C-14 gyro-stabilized compass system. Bearing pointers present automatic VOR bearing to station as selected. The situation display presents selected course, course deviation, and to/from ambiguity for VOR/localizer. The selected course is also shown on a digital display in the upper right portion of the HSI. Adjustments available on the HSI include a course set knob for selecting the desired VOR radial, and a heading knob for selecting a desired magnetic heading for flight director commands. A digital display in the upper left corner of the HSI presents DME distance. Failure mode flags are incorporated in the HSI display to indicate power failure, compass system failure, navigation information validity, and glideslope information validity.

The attitude direction indicator (ADI) is a multifunction indicator providing the pilot and copilot with basic flight situation and dynamic information and navigation command cues.

The ADI presents basic roll and pitch attitude information from the remote vertical displacement gyro. The ADI provides adjustments for roll and pitch trim. A turn rate indicator is incorporated into the lower bezel to provide information from the remote turn rate gyro. An inclinometer is housed above the turn rate indicator to provide slip-skid information. The ADI presents navigation command cues from the flight director computer on a vertical bar (roll cues), a horizontal command bar (pitch cues), and a longitudinal command pointer (third cue for collective/power adjustments). Glideslope deviations are presented on a pointer along the left edge of the bezel. Fail mode flags are available for vertical displacement gyro/attitude indicator failure, turn rate gyro failure, and flight director computer failure.

10.6 Landing Gear

The landing gear is a tricycle design with dual wheels on both the main and nose gears. Hydraulic actuators powered by the PC3 hydraulic system (described in section 10.2) retract the main gear into pods on the sides of the fuselage and the nose gear into the fuselage nose section. A 3000-psi nitrogen gas system is provided for emergency extension of the gear. Operation of the emergency control releases the uplocks and extends all gears to the down and locked position. Hydraulic brakes on each of the four main wheels are actuated by master cylinders, toe operated at the rudder pedals with mixing valves to prevent fluid transfer between pilot and copilot cylinders. The nose gear is full swiveling with shimmy damping and an automatic centering device that returns the gear to the trail position when unloaded. Deflection of the main-gear shock strut operates a switch which prevents inadvertent gear retraction and conversion to less than 60° pylon angle with the aircraft on the ground.

Normal landing gear operation is controlled by a landing-gear-position control handle mounted on the center console. Caution lights on the center instrument panel indicate nose gear "not centered," parking brake "on," and landing gear "up." An audible warning is sounded if the gear is not extended during flights below 200-ft radar altitude or 100-knot airspeed. A separate handle is provided to actuate the emergency landing gear extension system.

The landing gear is structurally designed to permit VTOL and STOL landings at design gross weight with a sink rate of 10 feet per second. Reserve energy sink rate is 12.25 feet per second at design gross weight. The maximum speed at which the gear may be extended or retracted is 160 knots.

10.7 Crew Station

10.7.1 *Cockpit arrangement* - The XV-15 crew station has side-by-side crew seating with LW-3B ejection seats that are described in section 10.8. The cockpit arrangement will permit the aircraft to be flown from either the pilot's seat on the right or the copilot's seat on the left. In addition to the instrument panel, the cockpit arrangement includes a center console, overhead console, and a right- and left- (pilot and copilot) side console.

10.7.1.1 *Instrument panel* - The pilot and copilot instrument panel displays are listed in table 10.7.1.1. The center section of the instrument panel presents system monitoring instrumentation that is in view of the pilot and copilot, but favors the pilot's view. This instrumentation consists of:

a) two gas producer tachometers (N_1)
b) two exhaust gas temperature indicators
c) two fuel flow indicators
d) two engine oil temperature and pressure indicators
e) two transmission oil temperature and pressure indicators
f) one dual dc voltmeter
g) one dual dc ammeter
h) one dual fuel quantity indicator
i) three hydraulic oil temperature and pressure indicators
j) one dual hydraulic quantity indicator for hydraulic system 1 and 2
k) landing gear position indicators
l) landing gear "up" warning indicator
m) time code generator display
n) 40-segment caution panel
o) differential trim indicator and differential control switch
p) transponder control panel

10.7.1.2 *Center console* - The center console equipment is arranged in two rows. The left side of the center console contains the following:

a) flight director control panel
b) stability and control augmentation control panel
c) force-feel control panel
d) rpm governor control panel
e) pilot's intercom system panel
f) parking brake handle
g) two throttles

The right side of the center console contains the following equipment:

a) UHF radio control head
b) UHF communication radio control head
c) VHF navigation radio control head

TABLE 10.7.1.1.- PILOT AND COPILOT INSTRUMENT PANEL DISPLAYS

	Pilot's panel	Copilot's panel
Encoding altimeter	X	
Barometric altimeter		X
Radar altimeter	X	
Airspeed indicator	X	X
Clock	X	
Attitude direction indicator	X	
Horizontal situation indicator	X	
Artificial horizon		X
Radio magnetic indicator		X
Triple tachometer (two engines plus rotor system)	X	X
Triple torquemeter (two rotors plus cross shaft)	X	X
Vertical speed indicator	X	X
Dual conversion angle indicator	X	X
Angle of attack indicator	X	X
Marker beacon lights (two)	X	
Accelerometer	X	
Flap position indicator	X	X
Master warning lights	X	X
Data control indicator		X
Rotor flapping indicator		X
Critical monitor meter		X
Critical temperature meter		X
Control position indicator	X	

d) rpm governor control wheel and indicator
e) pilot's collective-type power lever
f) landing gear handle
g) flap control handle

10.7.1.3 *Overhead console* - The overhead console contains the following equipment:

a) fuel control panel
b) hydraulic system control panel
c) electrical system control panel
d) lighting system control panel
e) engine anti-ice switch
f) pitot heat switch
g) electrical system circuit breakers

10.7.1.4 *Pilot's side console* - The pilot's side console (right side) contains the following equipment:

a) side window jettison control handle
b) pilot's seat height adjustment switch
c) compass control panel
d) pilot's oxygen control panel
e) map case
f) oxygen hose

10.7.1.5 *Copilot's side console* - The copilot's side console (left side) contains the following equipment:

a) copilot's collective-type power lever
b) copilot's seat height adjustment switch
c) environmental control unit panel
d) exciter control panel
e) copilot's intercom control panel
f) copilot's oxygen system control panel
g) map case
h) side window jettison handle
i) oxygen hose

10.8 Emergency Egress System

The emergency egress system includes the pilot and copilot zero-zero ejection seats and associated assemblies and hardware, and emergency release side and overhead panels.

The ejection seats are the flight qualified Rockwell International Model LW-3B seats. The ejection sequence must be initiated by each pilot independently as there is no interconnect between the two seats. Lateral trajectory divergence will be obtained by seat and track configuration to provide adequate separation during simultaneous ejection. Flexible oxygen and communication lines will be separated on ejection by pull-type breakaway fittings.

The overhead hatches and side windows for the pilot and copilot provide emergency release for ground egress. A lever is provided forward of the overhead console for a mild detonator cord release of the window glass of the upper hatch. The side windows may be released using a mild detonator cord attached to the inside window edge and activated by either pilot or copilot emergency release levers. An outside emergency release lever is also provided forward of the cockpit for ground release of the overhead and side window glass. All levers have safety pins and the outside emergency release lever is under an access cover. Emergency hatches and windows are identified by a broad, sharply contrasting band outlining the exit including a large arrow or other appropriate indication pointing to the release handle.

11. GROUND OPERATIONS AND SERVICING

11.1 Support Equipment and Systems

Utilization of the XV-15 aircraft will require various ground support items, tools, and equipment for inspection, rigging, testing, preparation for normal operation and routine maintenance of the aircraft. The above equipment can be grouped into two categories: (1) common equipment and (2) XV-15 peculiar equipment. Listings of the major items of these groups available for the XV-15 investigations are presented in tables 11.1.1 and 11.1.2.

TABLE 11.1.1.- COMMON SUPPORT EQUIPMENT

Class	Item	Equivalent Federal stock no. (FSN)	Use and components
Portable/mobile subsystem service carts			
	Generator set	6115-553-8957 type MD-3A	gasoline powered, supplies 28V dc (60 kW); 115V ac, 400 cps (45 kW)
	Ground air conditioner	4120-119-9384 type A-3	gasoline powered, supplies cooling air for crew station prior to a/c engine start
	Hydraulic system test stand	RX 4920-933-2823 5150 model AHT-63	electric motor powered, supplies 25 gal/min at 3000 psi, for ground testing a/c hydraulic systems

TABLE 11.1.1.- COMMON SUPPORT EQUIPMENT - Continued

Class	Item	Equivalent Federal stock no. (FSN)	Use and components
Portable/mobile subsystem service carts - Continued			
	Compressed air supply	4310-547-3741 type MC-2A	provides compressed air for pneumatic hand tools, servicing tires, pneumatic struts
	Oxygen service cart		servicing A/C oxygen system (up to 1800 psi)
	Nitrogen service cart		landing gear struts, and emergency nitrogen bottle
Work stands, jacks, platforms, and transportation equipment			
	Trailer	1730-852-0126	installation and removal of transmissions and engines (in conjunction with adapters - table 11.1.2)
	Transportation trailer	1740-516-7930 model 3000	transporting and maintenance platform for engines and transmissions
	Maintenance platform	1730-394-8883 type B-4A	aircraft maintenance/ servicing platform height 3 to 7 ft
	Maintenance platform	1730-395-2781 type C-1	aircraft maintenance platform height 4 ft (also use with type B-1)
	Maintenance platform	1730-390-5618 type B-1	heavy duty work platform, adjustable 3 to 10 ft

TABLE 11.1.1.- COMMON SUPPORT EQUIPMENT - Continued

Class	Item	Equivalent Federal stock no. (FSN)	Use and components
Work stands, jacks, platforms, and transportation equipment - continued			
	Hydraulic jack (2)	1730-516-2019 type B-6	10-ton lift capability, aircraft wing jack
	Hydraulic jack	1730-391-7932 type N-3T	aircraft nose and fuselage jack
	Towing tractor	3930-554-1534	move mobile ground equip., position aircraft
	Aircraft towbar	1730-967-9556	tow aircraft forward or push rearward, use on towing tractor
Measuring and testing equipment			
	Aircraft scale	6670-526-8497	determine aircraft weight and balance
	Multimeter	6625-242-5023	measure ac and dc electrical parameters
	Pitot-static testor	4920-474-8311	checks pitot-static system performance
	JET-CAL analyzer	4920-776-8756	Measures powerplant temp. indication system and components
	Liquid quantity testor	4920-509-1508 type MD-2A	checks fuel quantity indication system
	Battery charger/ analyzer		for servicing battery

TABLE 11.1.1.- COMMON SUPPORT EQUIPMENT - Concluded

Class	Item	Equivalent Federal stock no. (FSN)	Use and components
Miscellaneous service equipment			
	Electric rectifier power supply	6130-624-9099	in-hangar: used to supply electric power for aircraft systems calibration and checks
	Air injection pump	4320-390-9556	used with compressed air supply for servicing pneumatic struts
	Hydraulic dispenser	4920-245-1832	for servicing hydraulic system reservoirs
	Torque power wrench	5130-152-2082	hand crank torque multiplier used in removal of rotor retaining nut from wing pylon

TABLE 11.1.2.- XV-15 PECULIAR SUPPORT EQUIPMENT

Class	Item	Use and comments
Workstands, platforms, and transportation equipment		
	Transmission sling and adapter set	lifting, hoisting, transporting main transmissions
	Transmission stand adapter	mounting base for assembly or transportation of main transmission
	Rotor sling	lifting rotor from pylon mast
	Rotor hub support adapter	supports hub assembly or hub and blades - mounted on standard workbench
	Fuselage support	supports aft fuselage for stability during jacking operations
	Engine cradle	permits handling/transportation of T53 engine assembly
Rigging, measuring and testing equipment		
	Rotor assembly balance arbor and adapter	indicates and measures static balance of complete rotor assembly
	Universal joint centering fixture	for aligning and assembly of universal joints
	Control rigging kit	positioning, adjusting, tuning control linkages and surfaces
	Rpm governor test set	check operation of rpm governor circuitry

C-2

TABLE 11.1.2.- XV-15 PECULIAR SUPPORT EQUIPMENT - Concluded

Class	Item	Use and comments
Miscellaneous service and assembly equipment		
	Interconnect driveshaft tools	
	Rotor installation wrenches	
	Rotor blade positioning link set	to restrain rotor blades to prevent grip/grip seal damage during removal or installation of rotor
	Blade retention pin nut wrench	
	Blade bearing puller	
	Transmission & rotor assembly tool kit	
	Landing gear locking sleeves	to preclude inadvertent gear retraction while servicing aircraft on ground
	Window release lever safety pins	to preclude inadvertent firing while servicing aircraft
	Ejection seat safety pins	to preclude inadvertent firing while servicing aircraft
	Dust covers	turbine exhausts, turbine air inlets, pitot tubes, airspeed/attitude head
	Quill removal jackscrew set	for removing main transmission and interconnect gearbox quills
	Elevator and rudder positioning templates	for rigging elevator and rudder
	Seal installation plug kit	allows installation of Chevron-style oil seals without damage

11.2 Fluid Requirements

The fuel, oil, and hydraulic fluid required for the XV-15 aircraft is listed in table 11.2.1. Prior to servicing the aircraft with any fluids, the Inspection and Maintenance Manual, and aircraft-mounted decals and placards should be consulted.

TABLE 11.2.1.- XV-15 FLUIDS

Fuel	MIL-T-5624, JP-4 or JP-5
Engine Oil	MIL-L-23699, or MIL-L-7808
Transmission Oil	MIL-L-23699
Hydraulic Fluid	MIL-H-5606
Brake System Fluid	MIL-H-5606

11.3 Earth Grounding

The XV-15 aircraft will be grounded to earth during any ground service operation using one of the grounding jacks provided on the aircraft. Fueling operations will utilize grounding receptacles near each fuel filler opening and oxygen refilling operations will require the use of the grounding receptacle at the servicing point. A separate grounding receptacle near the external electrical power connector will be used at Ames Research Center for grounding to the service cart in addition to the ground in the three-prong connector. The hydraulic and air conditioning AGE are not separately grounded to the serviced aircraft under current Ames Research Center procedure.

11.4 Servicing

Trained aircraft service personnel will perform all maintenance, modifications, and inspections required for the flight research program. Inspection and maintenance manuals will be available and used as reference material in performing this work.

12. SUMMARY OF TILT ROTOR RESEARCH AIRCRAFT SAFETY FEATURES

The minimization of hazards and risks and the achievement of flight safety have been recognized objectives for the Tilt Rotor Research Aircraft Project since its inception. Even prior to the formal organization of the Project Office, the active developers of the tilt rotor concept in industry have been studying and developing concepts that provide for or enhance the safety of this V/STOL aircraft. Many of these points, and additional features, considered by the Project Office to be desirable or necessary for safety have been incorporated into the Tilt Rotor Research Aircraft.

In light of the significance of system safety to the success of this project, major design safety features are addressed in the XV-15 Tilt Rotor Research Aircraft Model Specification and associated contract documentation.

In general the XV-15 will meet or exceed the strength, design and construction requirements of the FAA tentative Airworthiness Standard for Powered Lift Transport Category Aircraft (Part XX), and other applicable government standards and specifications. An outline of the aircraft's significant inherent and built-in safety characteristics is presented in the following paragraphs. This section summarizes the safety characteristics discussed in sections 2 through 11 of this Manual.

12.1 Safety Characteristics

The two low disc loading rotors may be operated in the autorotation mode for reduced descent rate emergency landing in case of total loss of power. Adequate collective pitch range and rotor solidity (total blade planform) permit rotor speed control during descent and provide flare thrust to reduce rate-of-sink. The landing gear is designed to withstand a vertical sink rate of 12.25 feet per second at the design gross weight.

The rotors are driven by wingtip mounted turbine engines derived from existing engines of proven high reliability. An interconnecting shaft system between the rotors (cross-shafting) allows either engine to power both rotors in the event of an engine or high-speed gearing transmission failure. Driving each of the rotors independently is also possible in the case of a cross-shaft failure.

Overrunning clutches in the engine speed reduction gearing automatically disconnect a failed engine from the drive system, thus allowing the effective use of available power. Redundant transmission housing mounting lugs prevent a catastrophic result from a single bolt or lug failure. The drive system strength requirements allow for uneven power distribution (such as single engine operation) and maneuver or gust transient loads and torques. For normal operation, torque limitations will be placarded and are a pilot-control function.

Single engine performance, stability, and control are similar to two engine operation at low power settings because of the cross-shafting. Cruise and transition can be performed as normal but single engine hover is limited to low payload weights. The dynamics of the rotor/nacelle/wing will be demonstrated to be stable by operating the complete aircraft in the Ames 40- by 80-Foot Wind Tunnel prior to flight. While wind tunnel testing will be limited to speeds of around 200 knots, the test will be used to verify analytical predictions and, with small-scale model testing at simulated higher speeds, will establish a firm basis for extrapolating stability predictions into the higher speed range. Therefore, in achieving the desired dynamic characteristics over the entire operating range, the failure of the stability and control augmentation system will not lead to a catastrophic instability. Furthermore, aeroelastic stability shall be maintained with the loss of at least 20 percent of the wing stiffness due to structural failure.

The primary conversion (nacelle tilt) mechanism is provided with dual hydraulic actuation and redundant control subsystems to enable full range operation after any single failure. In the event of complete primary system failure, the nacelles can be converted by the use of the back-up hydraulic system powered actuator. A nacelle synchronization feature is also provided.

The hydraulic system consists of two primary subsystems (PC1 and PC2) for the flight controls and a flight-control backup hydraulic system (PC3), which normally powers the landing gear actuators. The design of the tandem flight control actuators prevents any one of the basic failure modes (i.e., the loss of fluid pressure to one side of the actuator and the sticking of valve spools) from incapacitating the operation of the actuator. For critical flight control components, such as the flaperon and the collective pitch actuators, a shuttle valve is incorporated to utilize hydraulic fluid from the backup subsystem (PC3) upon the loss of a PC2 fluid supply failure. The backup subsystem will also be automatically isolated from all non-essential components upon the loss of either PC1 or PC2. For this case, pneumatic backup is used to extend the landing gear.

Critical components of the three separate hydraulic systems will be physically isolated, where possible, to prevent concurrent failure due to local damage. The flight controls will be irreversible and include a force-feel and a stability and control augmentation system. The controls revert to an unboosted direct manual system in the event of hydraulic power loss on the hydraulic systems. Single actuators are used for controls that are not flight-critical items. Built-in test equipment will be provided. Standard aircraft hydraulic fluid will be used to reduce the fire potential of the hydraulic system. All of the hydraulic pumps are driven by the low speed drive train of the rotor transmission to assure hydraulic power as long as the rotors are rotating, in the event of failure of either or both engines.

Two independent primary electrical power generation, ac conversion, and distribution systems will be provided. Adequate electrical power for the critical flight required equipment will be available after the loss of one of the electrical systems. The electrical power for all equipment required during emergency autorotative flight (with both generators inoperative) can be provided from the battery source. The research instrumentation is electrically isolated from the flight instrumentation and flight control electrical components. Therefore, research instrumentation malfunctions will not adversely affect safety of flight. The rotor governing system will be provided with a failsafe capability. A third hydraulic power system, independent of the two systems used for flight control, is used for landing gear actuation. The landing gear, selected from a similar gross weight aircraft (CL-84), has already accomplished pertinent flight-worthiness tests. No special braking requirements for this V/STOL research vehicle are anticipated. The braking hydraulic system is separate from the aircraft hydraulic power systems.

The nose landing gear and primary supporting structure is located where, in the event of malfunction or structural failure, it will not present a direct hazard to the crew.

Crew safety is obtained by the design of the crew station (minimization of protruding rigid hazards and adequate crew restraints), and by provision of emergency egress systems (ejection seats and emergency release windows and hatches). Noninterfering, simultaneous ejections with zero-zero capability may be obtained with this ejection system.

An engine fire detection and pilot actuated fire extinguishing system will be incorporated. Engine inlet icing detection and anti-icing is also provided. Fuel is stored in the wings, outboard of the fuselage, in crash-worthy rip-resistant rubberized fabric cells. Breakaway fittings are utilized to eliminate fuel spillage from fuel lines separated in a crash. Furthermore, the wing structure and honeycomb/skin construction protects the fuel cells in low energy crash situations.

The remote location of the engines from the crew station reduce the hazard to the crew of engine fire and the resulting smoke and heat.

Nose gear swiveling and differential braking are provided for ground operation. The rotor disc plane in the VTOL takeoff configuration will be over ten feet above ground level at the design gross weight. The crew members will have a nearly unobstructed view of the outboard rotor tip path to reduce the hazard of rotor tip collision with ground objects during taxi or ground maneuvering.

Flight operations are expected to display safety characteristics similar to helicopters or conventional aircraft. High hover mode thrust available to weight ratios coupled with control powers and sensitivities greater than the minimum levels recommended in AGARD Report No. 577 will permit hover mode research, in and out of ground effect, with adequate control about all axes. Other available references to desirable handling qualities and stability and control characteristics for V/STOL aircraft, such as MIL-F-83300, MIL-F-8785B, and the tentative FAA standards entitled "Tentative Airworthiness Standards for Powered Lift Transport Category Aircraft," have been used in developing the V/STOL Tilt Rotor Research Aircraft.

Transition to cruise flight is performed within the boundaries established by wing stall, the torque limit, or rotor/hub endurance limits. The allowable corridor is broad (generally greater than 80 knots) and design criteria allow the fatigue limits to be penetrated briefly without incurring any dangerous control or structural problems. Control power equal to, or greater than, the recommendations of recent documentation (such as NASA TN-5594) will be provided about all axes throughout the transition process.

The general flight characteristics in cruise are those of a turboprop airplane. Conventional aircraft control surfaces are employed.

A pilot caution and warning system will provide visual and/or audible indications of detectable system malfunctions, such as hydraulic system pressure loss, rotor control discrepancies, gear up below 200 ft altitude, or 100 knots airspeed, engine fire, etc. Instrumentation will be incorporated to monitor loads and positions at critical locations (such as control linkages, control surfaces, etc.) during flight.

12.2 Flight Readiness

A significant factor enhancing the safety-of-flight of the Tilt Rotor Research Aircraft is the level of effort devoted to exploring the aircraft's functional and operational characteristics prior to the first flight. Thus, a program of model tests, component tests, a tie-down test, wind tunnel tests and real time pilot-in-the-loop flight simulation of the aircraft is included in the Project Plan.

Future investigations with the XV-15 aircraft should also utilize this broad background to assure the adequacy of technology and the level of XV-15 flight readiness for the new experiment.

13. XV-15 PROJECT DOCUMENTS

The pertinent documents developed in support of contract NAS2-7800 are listed below. Distribution of these documents is controlled by the XV-15 Tilt Rotor Research Aircraft Project Office, Ames Research Center.

	Document	Availability
1.	Model Specification	General (Government and Industry)
2.	NASA/Army Project Plan	Government
3.	Contractor Program Plan	General
4.	Contractor Test and Evaluation Plan	General
5.	Contractor System Safety Plan	General
6.	Project Risk Analysis	Government
7.	Flight Risk Analysis	Government
8.	Contractor Final Design Report*	General
9.	XV-15 Inspection and Maintenance Manual*	General
10.	XV-15 AGE Inspection and Maintenance Manual*	General
11.	XV-15 Flight Operations Manual*	General

*Documents noted by an asterisk are scheduled to be released later in the program.

12.	XV-15 Instrumentation and Data Acquisition System Manual*	General
13.	XV-15 Flight Test Plan*	General

*Documents noted by an asterisk are scheduled to be released later in the program.